CR-137789

26085-6001-TU00

DESIGN OF MULTI-MISSION CHEMICAL PROPULSION MODULES FOR PLANETARY ORBITERS

VOLUME I: SUMMARY REPORT

15 AUGUST 1975

Prepared for
NASA AMES RESEARCH CENTER
under
Contract NAS2-8370

TRW
SYSTEMS GROUP

(NASA-CR-137789) DESIGN OF MULTI-MISSION CHEMICAL PROPULSION MODULES FOR PLANETARY ORBITERS. VOLUME 1: SUMMARY REPORT (TRW Systems Group) 88 p HC $5.00 CSCL 21H

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ERRATA

"Design of Multi Mission Chemical Propulsion Modules for Planetary Orbiters," Report No. 26085-6001-TU-00 by TRW Systems Group, 15 August 1975.

Volume I, Summary

Page 56, Paragraph 5, Line 1: sentence should start as follows: "For LF_2/N_2H_4 systems ..."

Page 63, last line, last word: replace "tank" by "development".

Page 67, Paragraph 2, Line 6: replace "gain" by "gained".

Page 70, Paragraph 3, Line 1: replace "mission" by "missions"
Line 6: 5th word should read "awaiting".

Page 73, Table 12, Heading of Column 5: replace $|\Delta R| P_o$ by $|\Delta R|/R_o$.

Volume II, Technical Report

Page 3-14, Line 4: change "increases" to "decreases"
Line 5: change "increase" to "decrease".

Page 4-55, Table 4-7, first subheading: change "Module A" to "Module B".

Page 4-60, Figure 4-20: add symbol "C_2" to identify great circle through Z_o and S.

Page 4-63, Line 1 in text: change SLA to SPA.

Page 4-78, Equation (2): replace symbol t_m by t_M.

Page 4-83, Figure 4-28: replace symbol l_T by l_c.

Page 6-8, Paragraph 1, Line 8: change "low-emittance" to "low-absorptance".

Page 7-1, Paragraph 2, last line: change "Table 5-1" to "Table 4-1".

Page 7-11, 4th line from bottom: change "increases" to "decreases".

Page 7-12, Figure 7-8, legend in lower graph, right side: change "Space Tub" to "Space Tug".

Page 7-14, Table 7-3, legend of Column 6: change "Inert Mass" to "Total Inert Mass".

Page 7-17, Figure 7-10: dashed curve on lower right should be changed so as to depart from curve shown at $\Delta V = 3.5$ km/sec, intersecting solid curve at crossing of abscissa axis.

Page 8-3, last line: insert "prior" before "to payload integration".

Page 9-14, Table 9-3, heading of Column 5: replace $|\Delta R| P_o$ by $|\Delta R| / R_o$.

Page 9-15, equation in Paragraph 5: replace CE by CB_M.

Volume III, Appendixes

Page A-5, last term in Line 2 should read: LF_2/N_2H_4.

Page F-3, in figure caption: replace F-2 by F-1.

Page F-4, Table F-2: last line under "Assumptions" behind "Thrust Level" insert 600 lb_f.

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ABSTRACT

This report presents results of a conceptual design and feasibility study of chemical propulsion stages that can serve as modular propulsion units, with little or no modification, on a variety of planetary orbit missions, including orbiters of Mercury, Saturn, and Uranus. Planetary spacecraft of existing design or currently under development, viz., spacecraft of the Pioneer and Mariner families, are assumed as payload vehicles. Thus, operating requirements of spin-stabilized and 3-axis stabilized spacecraft have to be met by the respective propulsion module designs. As launch vehicle for these missions (considered for the mid-1980's or thereafter) the Shuttle orbiter and interplanetary injection stage, or Tug, plus solid-propellant kick motor was assumed. Accommodation constraints and interfaces involving the payloads and the launch vehicle are considered in the propulsion module design.

In this 12-month study TRW evaluated the applicability and performance advantages of the space-storable high-energy bipropellants (liquid fluorine/hydrazine) as alternative to earth-storable bipropellants (nitrogen tetroxide/monomethyl hydrazine). The incentive for using this advanced propulsion technology on planetary missions is the much greater performance potential when orbit insertion velocities in excess of 4 km/sec are required, as in the Mercury orbiter. Possible applications also include ballistic comet rendezvous missions. A major part of the study effort was devoted to design analyses and performance tradeoffs regarding earth-storable versus space-storable propulsion systems, and to assess cost and development schedules of multi-mission versus custom-designed propulsion modules. The report includes recommendations as to future research and development objectives in this field.

1. INTRODUCTION AND SCOPE OF STUDY

1.1 INTRODUCTION

Planetary exploration by orbiting spacecraft will be achievable at reduced cost by introducing a modular system concept. This requires development of advanced chemical propulsion stages suitable for use with existing planetary spacecraft designs such as Pioneer (spin-stabilized) or Mariner (three-axis stabilized). The propulsion modules are to be used in multiple mission applications, either for outer-planet or Mercury orbit missions.

In addition to exploring the feasibility of developing multi-mission propulsion modules for spinning or nonspinning spacecraft classes this study considered the use of space-storable versus earth-storable bipropellants in these modules. Space-storable bipropellants (fluorine/hydrazine) with a specific impulse (I_{sp}) as large as 375 seconds would increase the performance potential of the multi-mission propulsion module significantly beyond that of the conventional earth-storable bipropellants (nitrogen tetroxide/monomethyl hydrazine) with I_{sp} of about 295 seconds which were used by the Mariner 9 Mars orbiter mission (1971). However, use of the novel propulsion system, not yet developed for flight application, raises technology problems that were addressed in this study. Performance evaluations comparing the effectiveness of space-storable and earth-storable propellants in the multi-mission module for planetary orbiters also were a major study objective.

Flight time to the outer planets was a principal concern in the performance evaluations. The performance improvement achievable by the advanced space-storable bipropellant system is a major incentive in developing this new technology for flight use. Outer-planet orbiter missions beyond Jupiter become attractive and feasible only if mission times do not exceed the expected life times of components and subsystems of the spacecraft that are vital to the success of the mission. The greatest part of the mission is spent in transit from earth to the planet. Reduction of flight times involves 1) larger injection energies at earth and 2) increased arrival velocities at the planet. The first requirement reduces the total mass that can be injected into the heliocentric

trajectory to the planet by a given launch vehicle. The second requirement implies an increase in the mass of the retro-propulsion system used for orbit insertion at the target planet and, hence, an increase in the total injected mass for a given payload and designated orbit.

A third factor of major concern in this study was the feasibility of launching the planetary orbiter using the Shuttle/Upper Stage as launch vehicle, since none of the missions considered would be flown before the mid-1980's. In addition to performance, safety considerations of the Shuttle carrying a fluorinated propulsion system in its payload are a factor in establishing mission feasibility.

1.2 STUDY OBJECTIVES

The principal study objectives are the following:

1) To develop a conceptual design for each of four multi-mission chemical propulsion modules (two propellant combinations, two sizes), and to assess the capability of each in a number of missions requiring major midcourse and terminal propulsion maneuvers.

2) To assess the recurring and nonrecurring cost of these modules as function of the number of missions they might serve, and to estimate total time and cost required to develop the modules to operational status.

3) To identify and assess the cost-effectiveness of the new technology to be developed in order to meet design requirements most effectively.

Design activities and analyses performed in the study were subject to the following requirements and guidelines:

- A common propulsion module is required that can be used practically without modification, in different planetary orbit missions, namely, Mercury, Saturn, and Uranus orbiters. (The multi-mission module must be able to withstand the environmental extremes of missions close to the sun and at great distances from the sun.)

- Propulsion module designs are required for a) spin-stabilized payload vehicles of the Pioneer class, and b) three-axis stabilized vehicles of the Mariner class.

- The space-storable propulsion modules are to be compared with earth-storable modules of equivalent performance

- All missions are to use the Shuttle orbiter and an expendable upper stage as launch vehicle. Compatibility with Shuttle launch conditions and orbital operations must be assured.

The approach suggested to achieve the desired multi-mission commonality is to design a module with sufficient propellant capacity for intermediate impulse requirements, e.g., the Saturn orbit mission. The much greater impulse requirement of the Mercury orbit mission is met by using two propulsion modules in tandem. This not only avoids the weight penalty of overly large tank sizes and over 50 percent off-loading for the lower energy missions (with attendant propellant sloshing problems), but also yields a major performance improvement through two-stage orbit insertion at Mercury.

Figure 1 illustrates schematically the different payload vehicle classes and mission classes to which the propulsion modules will be applied. The propulsion stage designed for use with spin-stabilized (Pioneer type) payloads will be termed Module A; the stage designed for three-axis stabilized payloads (Mariner type) will be termed

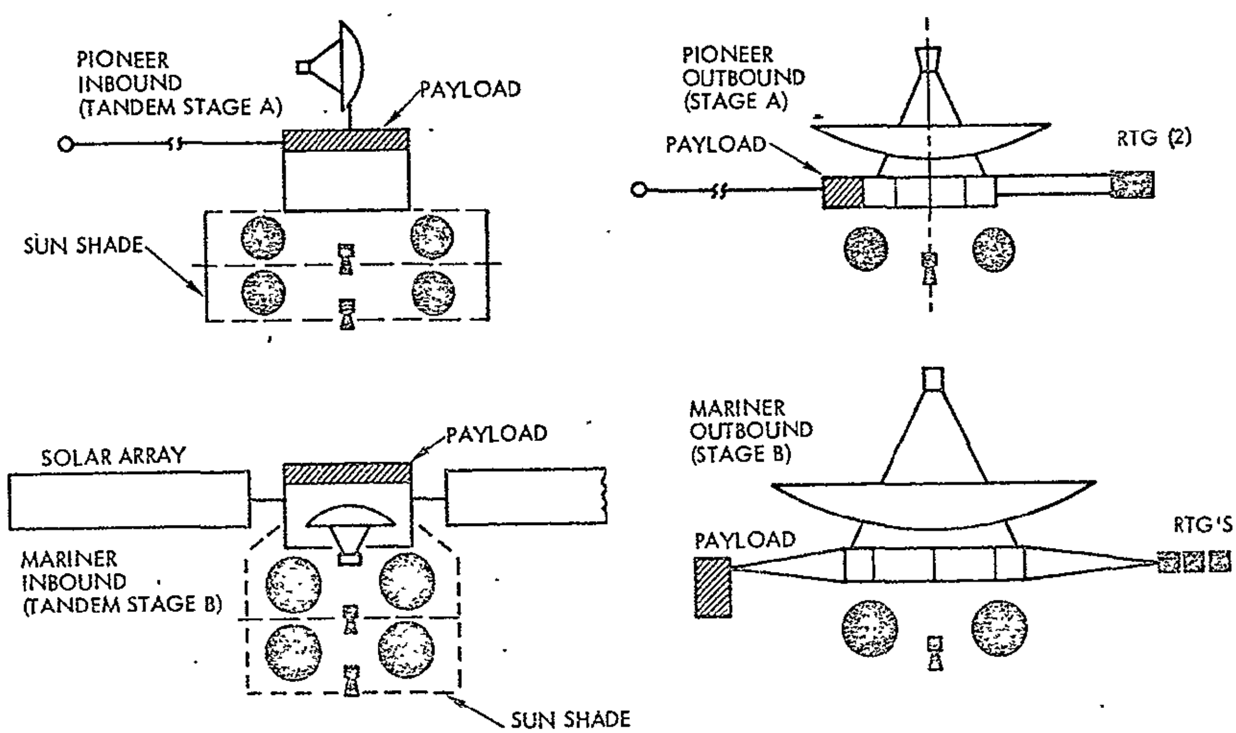


Figure 1. Specified Payload Spacecraft Configuration (Schematic)

Module B. The illustrations in Figure 1 show one-stage and tandem-stage arrangements for the outer planet ("outbound") and Mercury orbiter ("inbound") missions. In the inbound mission a sun shade is required to protect the Pioneer propulsion module and, in the Mariner case, the payload spacecraft against the intense solar radiation. Protection against intensive heat radiation from the dayside of Mercury must also be provided.

1.3 MISSION PERFORMANCE REQUIREMENTS

Primary missions to be performed by the multi-mission propulsion module are planetary orbit missions to Mercury (1988), Saturn (1985), and Uranus (1985). Rendezvous missions to the comets Tempel 2 (1983 and 1984), Faye (1986), Kopff (1991), Perinne-Mrkos (1990 and 1991), and Encke (1987), may also be within the capability of the multi-mission propulsion module, but are to be considered only as secondary objectives.

These missions have the common requirement for high impulsive energy but have very dissimilar characteristics otherwise: they require transit times ranging from 2 to 8 years or longer, are exposed to extremely different physical environments at solar distances ranging from 0.3 to 20 AU, and vary greatly in utilization of propulsive capabilities and thrust phase sequences. Figure 2 represents typical thrust phase sequences interrupted by long periods of dormancy.

Both Saturn and Uranus missions must use direct transfer trajectories, since a Jupiter swingby maneuver would lead to high arrival velocities and, hence, excessive orbit insertion velocity requirements. Transfer times are therefore quite long and, in some instances, can approach the duration of a Hohmann transfer.

Mission analysis, as such, was not included in the scope of TRW's study tasks. A considerable amount of related mission analysis work has been conducted by NASA, Ames Research Center, to define mission profile data and propulsion requirements that were furnished to TRW during this study. Propulsive requirements of all missions considered are summarized in Figure 3.

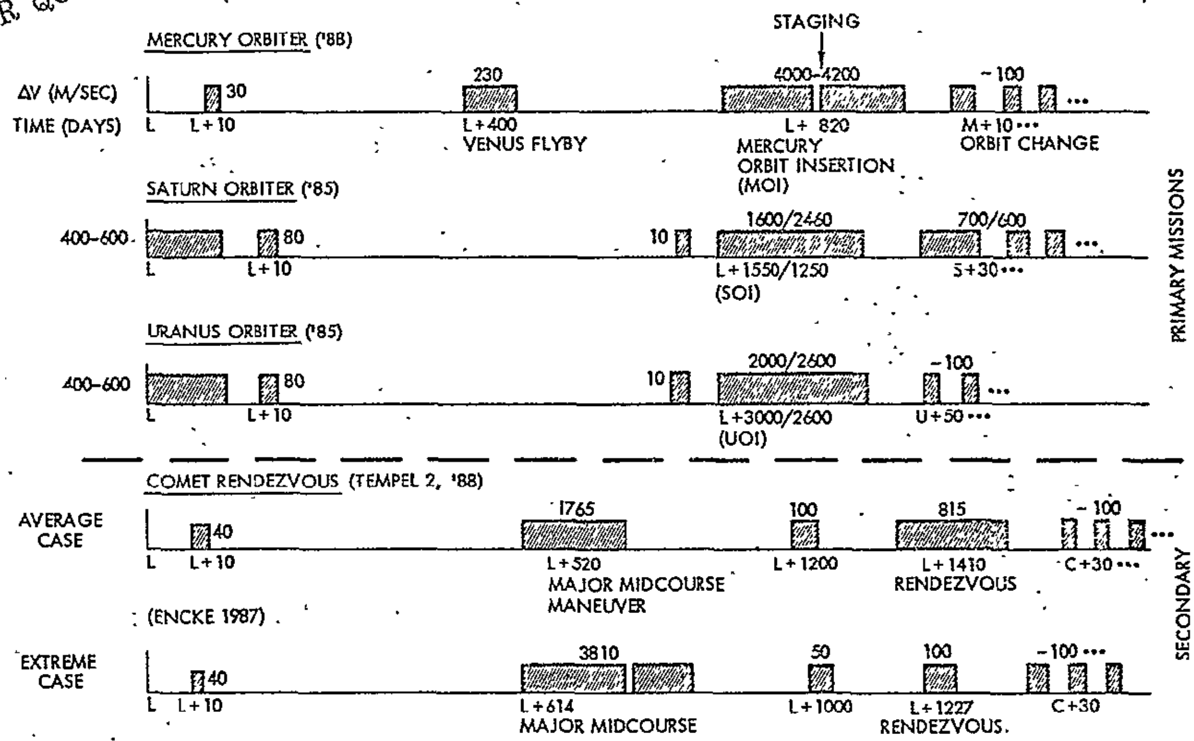


Figure 2. Propulsion Schedules (Preliminary); ΔV Impulses in m/sec

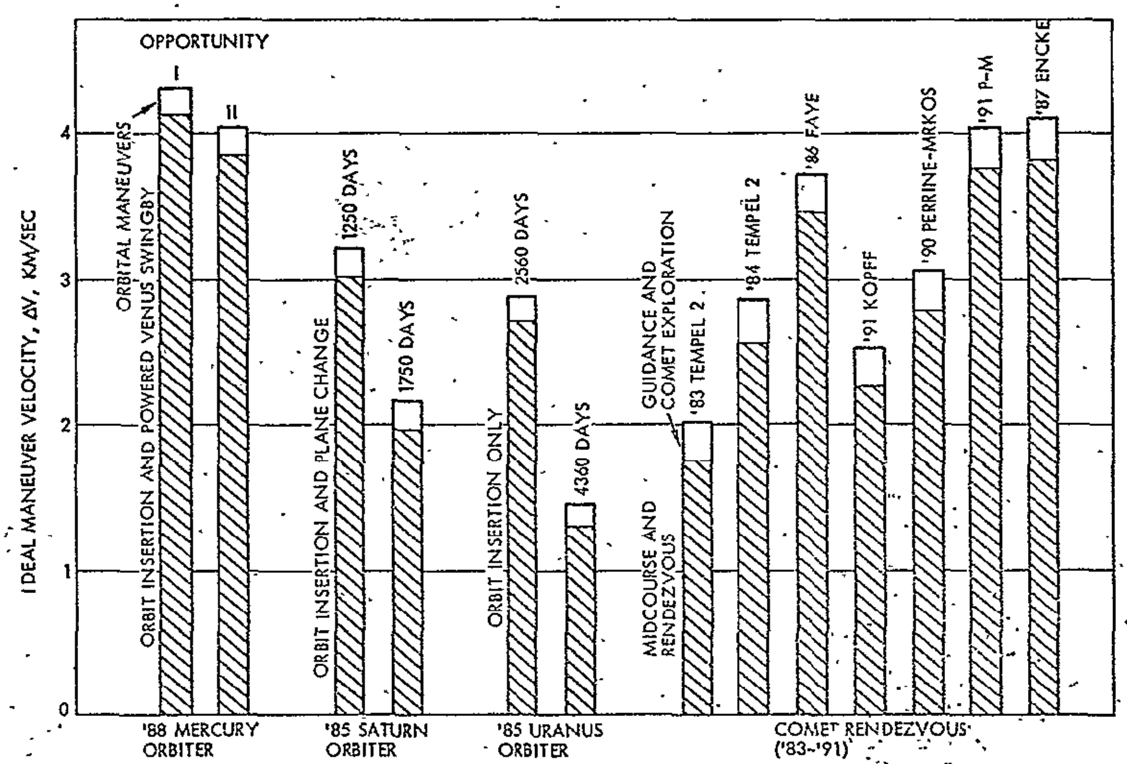


Figure 3. Ideal Maneuver Velocity Requirements for Specified Missions

1.4 IMPLICATIONS OF SHUTTLE LAUNCH

Accommodation of the flight spacecraft (which consists of the payload vehicle and one or two propulsion modules) on the Shuttle orbiter is a basic design and operational constraint. Figure 4 illustrates the Shuttle upper stage and flight spacecraft in stowed and extended configuration. The operational sequence after separation from the Shuttle orbiter includes: orientation of the upper stage preparatory to ignition; interplanetary trajectory injection of the flight spacecraft by the upper stage and by a solid kick motor (in the outer-planet mission). An initial thrust phase of the spacecraft propulsion module to augment launch vehicle performance may also be included (see below).

Payload accommodation requirements and constraints are defined in the Shuttle Payload Users Handbook (Reference 1) and Shuttle upper stage capabilities and configurations defined by NASA (Reference 2) and were taken into consideration in the propulsion module design.

The structural load profile for payloads carried by the Shuttle orbiter (from Reference 1) is defined in Table 1. Among constraints imposed by the Shuttle on the flight spacecraft is the contingency of a mission abort and return of the flight spacecraft from orbit. The highest structural loads occur during this abort mode with possible crash landing accelerations of up to 9 g's in axial and 4.5 g's in lateral direction. Other implications of an abort mode involve the disposal of payload propellants prior to abort initiation, to reduce the total cargo weight, and to avoid safety hazards from the load of hypergolic, toxic and corrosive propellants carried in the propulsion module.

Only a limited range of Shuttle interface and operational requirements could be addressed within the scope of this study. Extensive use was made of results obtained in previous and concurrent JPL studies of Shuttle-launched Mariner orbiters. Secondly, safety implications involved in the use of fluorinated bipropellants were investigated concurrently in a separate study performed by TRW under JPL contract. Results of that study are reflected in the propulsion module design.

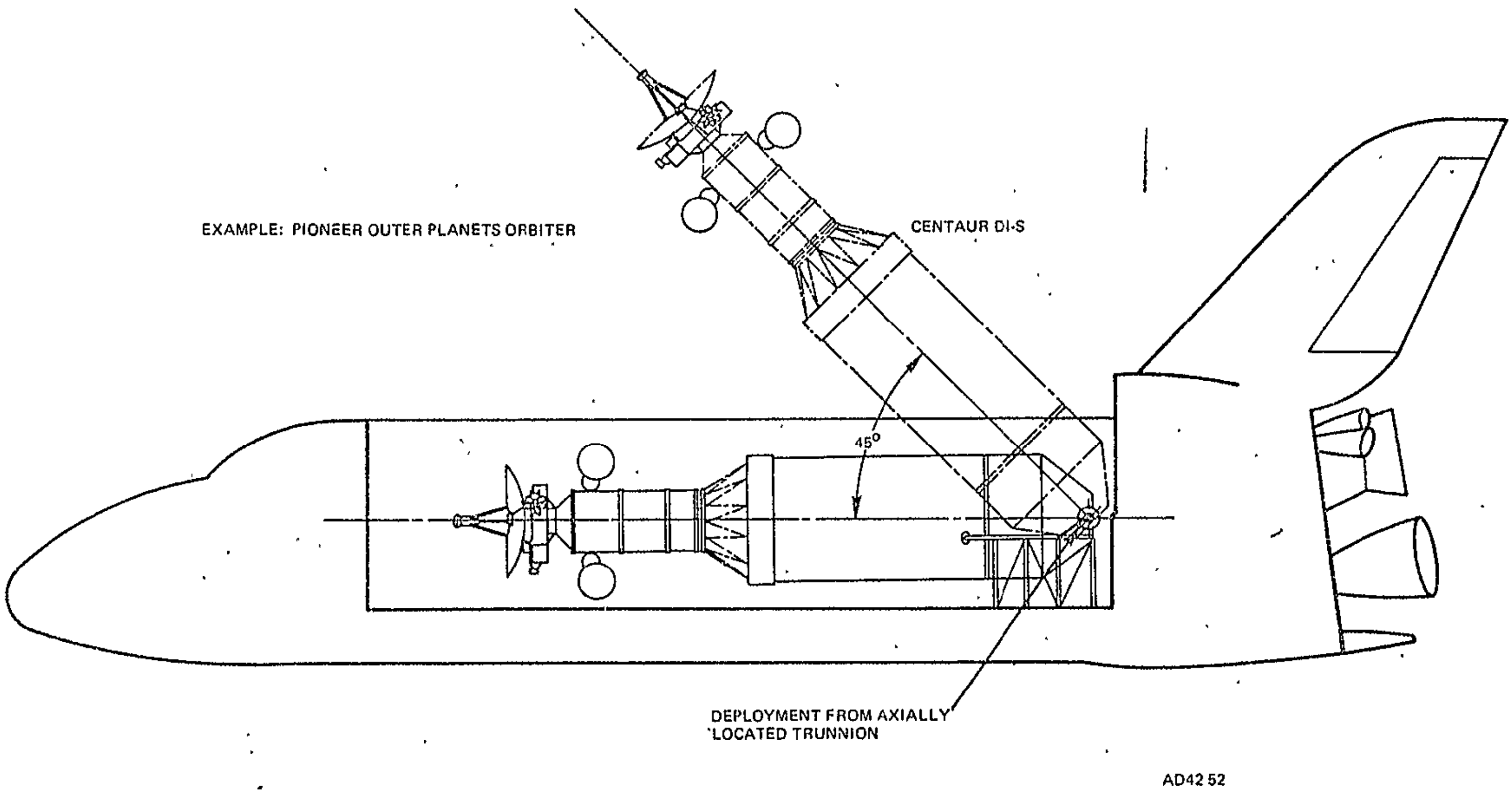


Figure 4. Deployment Procedure of Shuttle Upper Stage with Pioneer Orbiter

Table 1. Shuttle Payload Maximum Design Accelerations (g's)

Condition	Upper Stage Loading	a_x	a_y	a_z
Lift-Off	Full	-2.9	±1.0	±1.5
High Q boost	Full	-2.0	±0.5	±0.6
Booster end burn	Full	-3.3	±0.2	-0.75
Orbital operation	Full	-0.2	±0.1	±0.1
Entry and descent	Empty	0.75	±1.25	1.0
Landing	Empty	+1.0	±0.5	2.8
Crash (ultimate load)	Empty	9.0	±1.5	-4.5

Sign convention:
+x forward
+y left
+z upward

1.5 RELATED STUDIES

This study relates to and draws on previous work involving the use of space-storable bipropellant systems, primarily studies performed by JPL (References 3, 4, and 5).

A previous study performed in 1972 by TRW under JPL contract defined the thermal control methodology for fluorinated bipropellents and planetary orbiters (Reference 6). The results were applied to the multi-mission propulsion module design.

Several other studies performed at JPL, NASA/Ames, and TRW have defined design and performance characteristics of planetary orbiters of the Pioneer and Mariner class (References 7, 8 and 9), some of which reflect design requirements imposed by the use of Shuttle as launch vehicle.

TRW's concurrent study (Reference 10) developed methods for achieving a high level of safety in handling liquid fluorine prior to and during Shuttle launch. These results are directly applicable to and were utilized in formulation of the propulsion module design and handling concepts in the present study.

Performance evaluations of this study were augmented by results from a concurrent NASA, Ames Research Center, study by Duane W. Dugan (Reference 11). Data from Mercury orbiter mission studies by Martin Marietta (Reference 12) were also utilized.

2. PROPULSION MODULE CONFIGURATIONS

2.1 DESIGN APPROACH

2.1.1 Propulsion-Module Sizing

The preferred approach is to use two propulsion modules of equal size, arranged in tandem, for two-stage orbit insertion at Mercury. This provides:

- More weight-effective orbit insertion at Mercury
- Reduced inert weight of the propulsion module and, hence improved performance of the multi-mission module in outer-planet orbit missions
- Reduced ullage and, hence, reduced propellant sloshing in the outer planet mission.

Depending on thrust acclerations used and specific impulse of the propulsion system, the reduction in total propellant mass can be as large as 2:1 for the Mariner orbiter. Reduction of the propulsion module inert weight is correspondingly large.

In those cases where the multi-mission propulsion module, sized for the Mercury orbit mission, has more propellant capacity than required for outer-planet orbiters, the extra propellant can be utilized to augment launch vehicle performance. This requires an initial maneuver immediately after burnout of the Shuttle orbiter's solid propulsion motor. A delay of even a few minutes in spacecraft propulsion module ignition can reduce the desired C_3 augmentation up to 50 percent. Analysis shows that appreciable performance improvements, i.e., flight time reduction to Saturn or Uranus, are achievable by this maneuver mode only if space-storable propellants are used.

2.1.2 Mass Properties Control

Center-of-mass locations and moments of inertia of the payload spacecraft are fundamentally changed by addition of the large propulsion modules. This is of concern primarily in the case of the spin-stabilized system (Module A). To avoid unfavorable moment-of-inertia ratios it is

required in this case to spread the propellant tanks as far outward as possible within Shuttle cargo bay dimensions. In this manner it is possible to achieve spin moments of inertias at least 1.1 times greater than the maximum transverse moments of inertia, and thereby to insure long-term spin stability in all mission phases. The use of at least four tanks (two for oxidizer, two for fuel) is essential in the spinning configuration for proper mass balance.

Small residual center-of-mass deviations from the geometrical centerline and small thrust axis misalignments tend to produce nutations in spinning spacecraft during the thrust phase. The maximum nutation angle can be held to within about 1 degree by increasing the spin rate of the flight spacecraft prior to each thrust phase. Typical rate increases to three times the nominal values of 5 rpm and 10 rpm are envisioned for the outbound and inbound Pioneer applications, respectively. The increased spin rate also increases structural stiffness of deployed appendages against bending due to axial thrust acceleration.

2.1.3 Thrust-Level Selection

Thrust-level selection involves a trade between orbit insertion performance gains attained by high thrust acceleration on one hand due to large gravity losses at Mercury, versus weight penalties and payload spacecraft redesign requirements due to load on deployed appendages on the other. Propellant requirements for Mercury orbit insertion are very sensitive to thrust level (Figure 5). Those for orbit insertion at Saturn and Uranus are affected much less severely. Weight penalties that accrue from a thrust level increase include those associated with thruster size and those involving structural strengthening of payload appendages.

In the inbound Mariner spacecraft missions large thrust accelerations affect primarily the deployed solar panels. As originally designed they cannot withstand accelerations exceeding 0.01 g. Thrust levels required for effective Mercury orbit insertion are at least 10 times larger. The problem can be resolved by using guy wire to support the deployed panels.

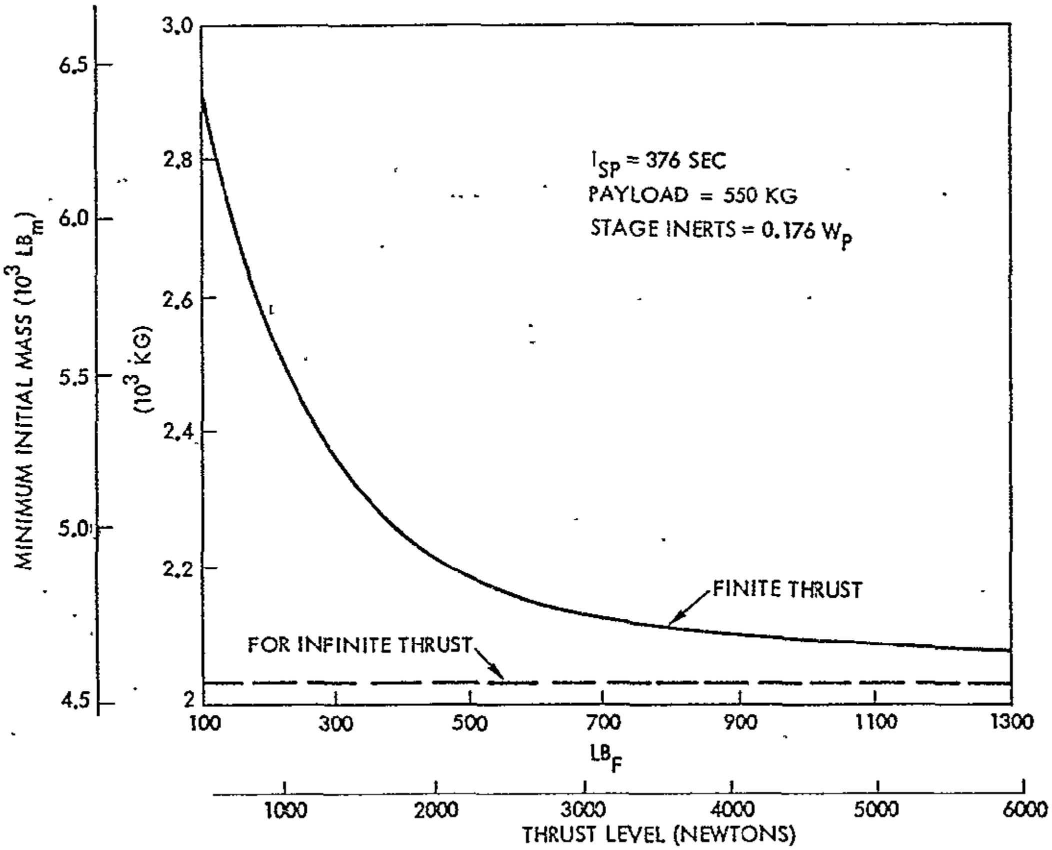


Figure 5. Minimum Initial Mass Versus Thrust Level for Mercury Orbiter (Tandem Stages)

The outbound Pioneer spacecraft also cannot withstand large thrust accelerations without a redesign of the RTG support arms and the magnetometer boom. Table 2 summarizes factors influencing the choice of thrust acceleration in the four spacecraft classes being considered. An acceptable compromise is achieved by changing the engine size from 800 lb_f (3560 N) for the Mercury orbiter to 200 lb_f (890 N) for outer-planet orbiters. The impact of this modification on design commonality can be minimized if the propellant feed system is designed to accommodate the propellant flow rate occurring with the larger size engine, and if some engine assembly elements such as valve assemblies remain unchanged. The plumbing, which remains the same, will constitute a small weight penalty for the smaller engine size.

Table 2. Principal Thrust Level Selection Constraints

Mission Constraints	Module A Payload (Spinning)	Module B Payload (Nonspinning)
Mercury Orbiter: Incurs major performance penalty for low thrust level	• Pioneer Venus orbiter class • Designed to withstand up to 8 g thrust acceleration (solid motor) in Venus orbit mission • Can readily accommodate desired large thrust level (600 to 800 pounds)	• Mariner Venus/Mercury flyby class • Deployed solar panels designed for accelerations ≤0.01 g • Solar panel support must be redesigned in any case to accommodate orbit insertion thrust
Outer Planet Orbiters: Can accept low thrust level with small performance loss	• Pioneer 10 and 11 Jupiter flyby class • Designed to withstand only up to 0.1 g thrust acceleration • Retraction of RTG and experiment booms prior to thrust initiation impractical • Minor redesign can accommodate up to 0.2 g. Acceleration level >0.4 g requires more significant design changes	• Mariner Jupiter/Saturn flyby class • Instrument and RTG support arms can tolerate up to 0.2 g • Long experiment booms can be retracted prior to thrust initiation

2.1.4 Auxiliary-Propulsion Functions

Redesign of the auxiliary propulsion system of the payload vehicles is necessitated by the attachment of the propulsion module at the aft end. If the propulsion module is retained during the orbital phase it can be used to support auxiliary propulsion functions (orbit corrections as well as attitude control maneuvers) in addition to performing the primary high thrust maneuvers. This has the following advantages:

- A common propellant supply is used
- The auxiliary system operates on regulated pressure rather than in the blowdown mode, with higher average specific impulse.
- In some applications the auxiliary propulsion system can utilize bipropellants rather than monopropellant hydrazine at higher specific impulse
- Allocation of hydrazine from the common propellant supply (in the case of space-storable propellants) for use by auxiliary thrusters is consistent with a favorable mixture ratio for the bipropellant main engine

- Placement of auxiliary thrusters on the propulsion module rather than on the payload vehicle reduces unwanted inter-axis coupling torques
- Integration of main propulsion and auxiliary propulsion into the propulsion module simplifies assembly and test operations and reduces cost.

These considerations apply except for the case of propulsion Module B with earth-storable propellants. The minimum impulse bit required for effective limit cycle attitude control is an order of magnitude smaller than that achievable by bipropellant (N_2O_4/MMH) auxiliary thrusters, and therefore a separate monopropellant supply is required. The hydrazine tank in the Mariner payload vehicle will be utilized for this purpose. Table 3 summarizes the preferred auxiliary propulsion design approach for the different payloads and propulsion systems being considered.

Combining auxiliary propulsion with the main propulsion module function is a practical approach only if the propulsion module is retained during the orbiter phase. Relative advantages and disadvantages of this option are summarized in Table 4.

Performance advantages resulting from the use of a common propellant supply for primary and auxiliary propulsion functions, as such, do not provide a sufficient argument for propulsion module retention. However, retention is justified because of the greater maneuvering reserve it offers for mission flexibility under unknown environmental conditions at destination, for greater scientific mission yield as well as for spacecraft protection against unforeseen hazards.

2.1.5 Selection Rationale for Propulsion Module A and B Design

The selected configurations for propulsion Modules A and B (see next section) evolved as a result of tradeoffs and practical design preferences, subject to the following requirements and constraints:

- Structural requirements
- Thermal requirements
- Attitude control and dynamics constraints.

Table 3. Auxiliary Propulsion Implementation

Propulsion Module Type	Propellants for Main Engine	Force lb_f (Newton)	Type	Minimum Thrust Pulse (sec)	Pressure (psi)	I_{sp} (sec)	Remarks
A	N_2O_4/MMH	2 to 5 (9.1 to 22.3)	Bipropellant N_2O_4/MMH	0.5	300 (regulated)	260-280	• Bipropellant thrusters most efficient; within state of technology
	F_2/N_2H_4	1 to 2 (4.5 to 9.1)	Monopropellant N_2H_4	0.03	300 (regulated)	180-220	• Uses spare fuel tank capacity (provided to improve main engine mixture ratio)
B	N_2O_4/MMH	0.3 to 0.5 (1.4 to 2.3)	Monopropellant N_2H_4	0.03	150-300 (blowdown)	170-220	• Requires separate hydrazine tank(s) on payload spacecraft (bipropellant thrusters would have larger than acceptable minimum impulse bit)
	F_2/N_2H_4	0.3 to 0.5 (1.4 to 2.3)	Monopropellant N_2H_4	0.03	300 (regulated)	180-220	• Uses spare fuel tank capacity

Notes:

- All but Module B (earth storable) use auxiliary propellant from own propellant supply
- Auxiliary thrusters on propulsion module in all cases
- Optimum mixture ratio (1.5:1) in space-storable case implies extra fuel tank capacity if equal volume tanks are used

Table 4. System Considerations Regarding Propulsion Module Retention

Advantages	Disadvantages
Retention of Propulsion Module	
• Increased flexibility of orbital phase	• Exposure of propulsion module to increased meteoroid impact hazard
• Ability to make significant orbit trims late in the mission (desirable from scientific standpoint)	• Extended electrical power requirement for propulsion module heating
• Ability to use main propellant supply for auxiliary propulsion (performance gain)	
• Propulsion module shields spacecraft rear side against meteoroids	
Staging of Propulsion Module	
• Reduced mass and moments of inertia improve auxiliary propulsion performance	• Possible malfunction of pyrotechnic separation devices after long transit time introduces failure mode
• Elimination of some science instrument and antenna field-of-view obstructions	• Required increase in payload spacecraft propulsion capability in lieu of using spare propellant capacity of the propulsion module could be potentially costly (tank size)

- Weight conservation
- Multi-mission commonality constraints
- Payload accommodation and interfaces
- Shuttle/upper stage accommodation and interfaces.

Figures 6 and 7 summarize the selection of major propulsion module configuration features and give the rationale used in making these selections.

2.2 CONFIGURATION OF MODULE A (SPIN-STABILIZED PAYLOAD)

2.2.1 Mercury Orbiter

Figure 8 shows the selected propulsion module design for spin-stabilized payloads arranged in tandem for the Mercury orbit mission.

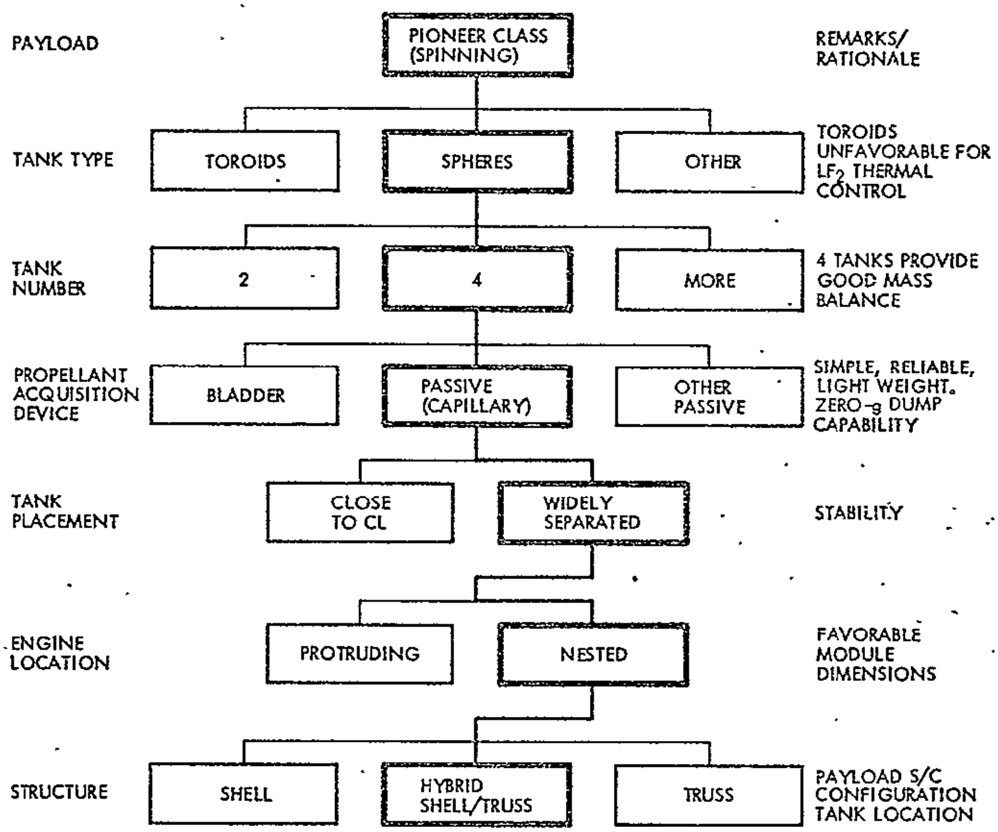


Figure 6. Preferred Design Options for Propulsion Module A

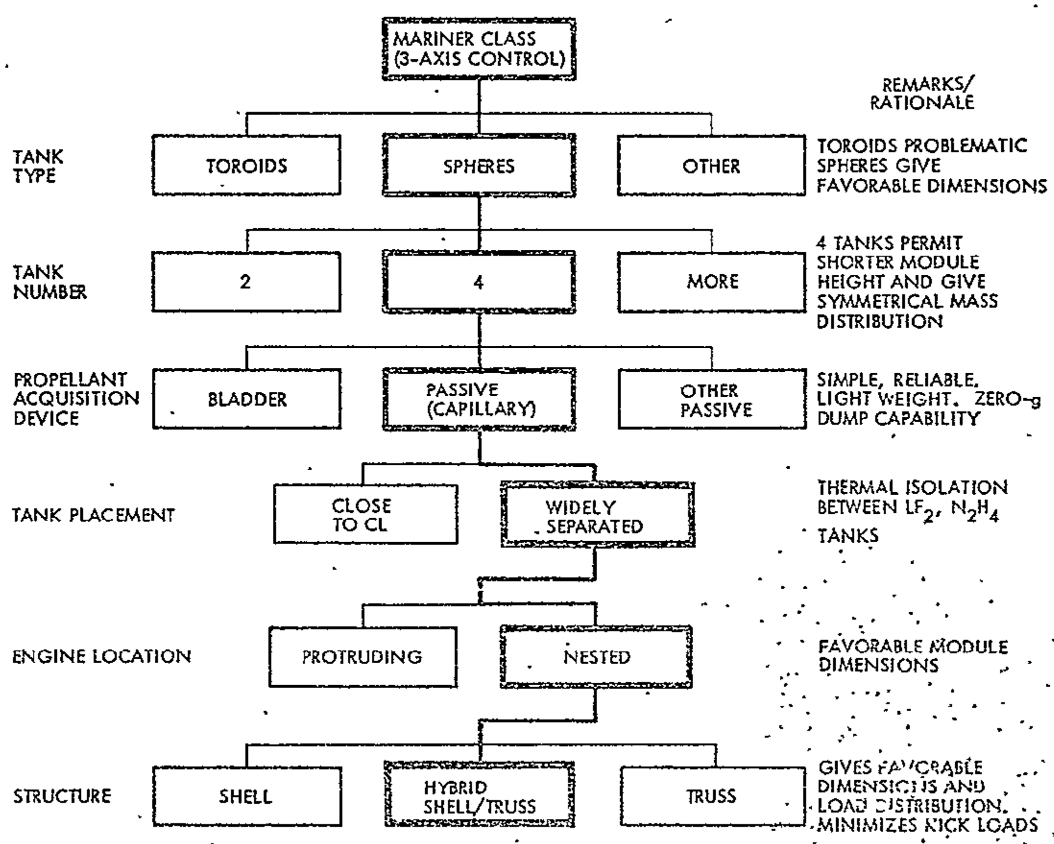
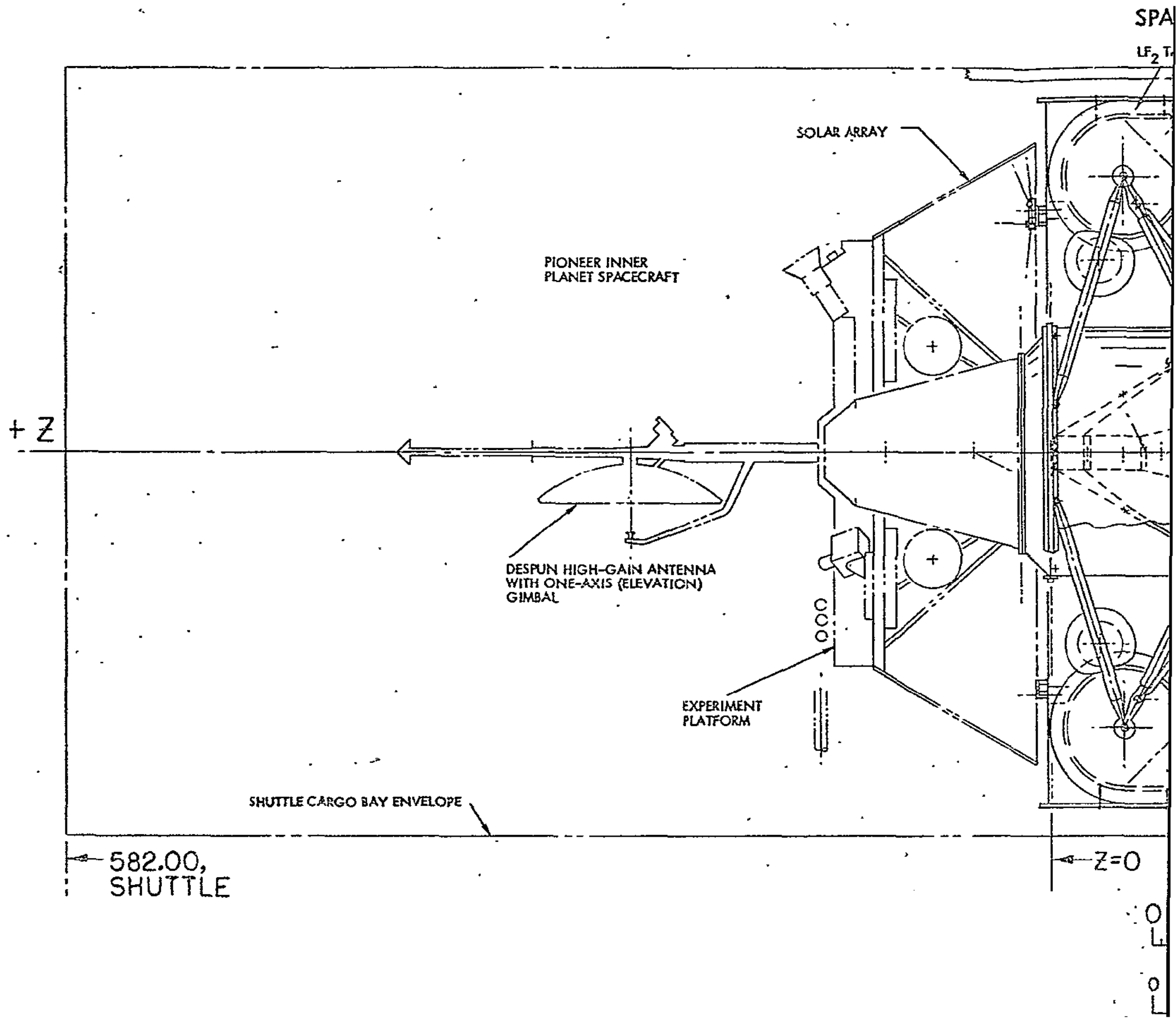


Figure 7. Preferred Design Options for Propulsion Module B


SPA
SOLAR ARRAY
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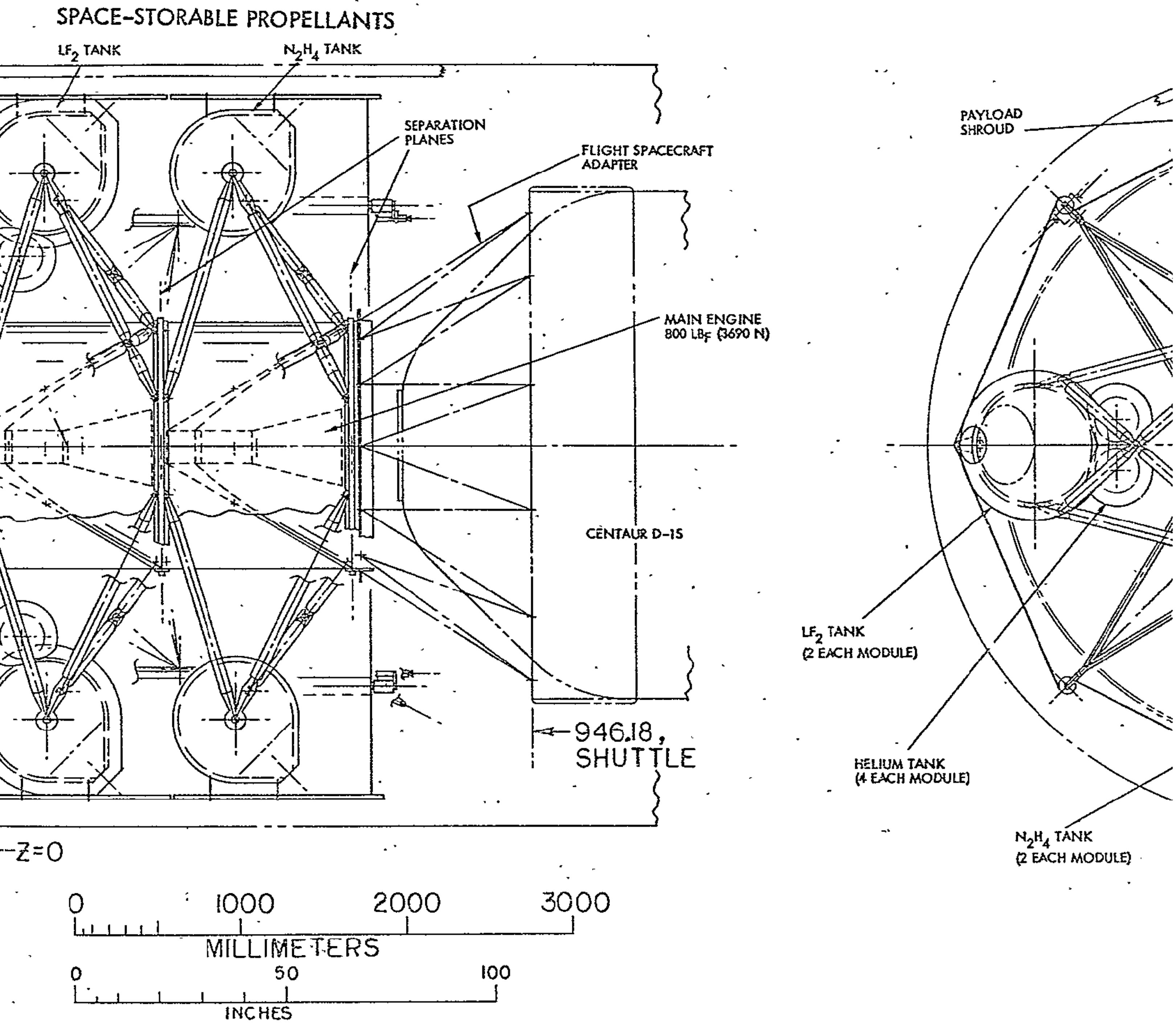
SPACE-STORABLE PROPELLANTS
LF_2 TANK
N_2H_4 TANK
SEPARATION PLANES
FLIGHT SPACECRAFT ADAPTER
MAIN ENGINE 800 LBF (3690 N)
CENTAUR D-1S
946.18, SHUTTLE
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0 1000 2000 3000
MILLIMETERS
0 50 100
INCHES
PAYLOAD SHROUD
LF_2 TANK (2 EACH MODULE)
HELIUM TANK (4 EACH MODULE)
N_2H_4 TANK (2 EACH MODULE)

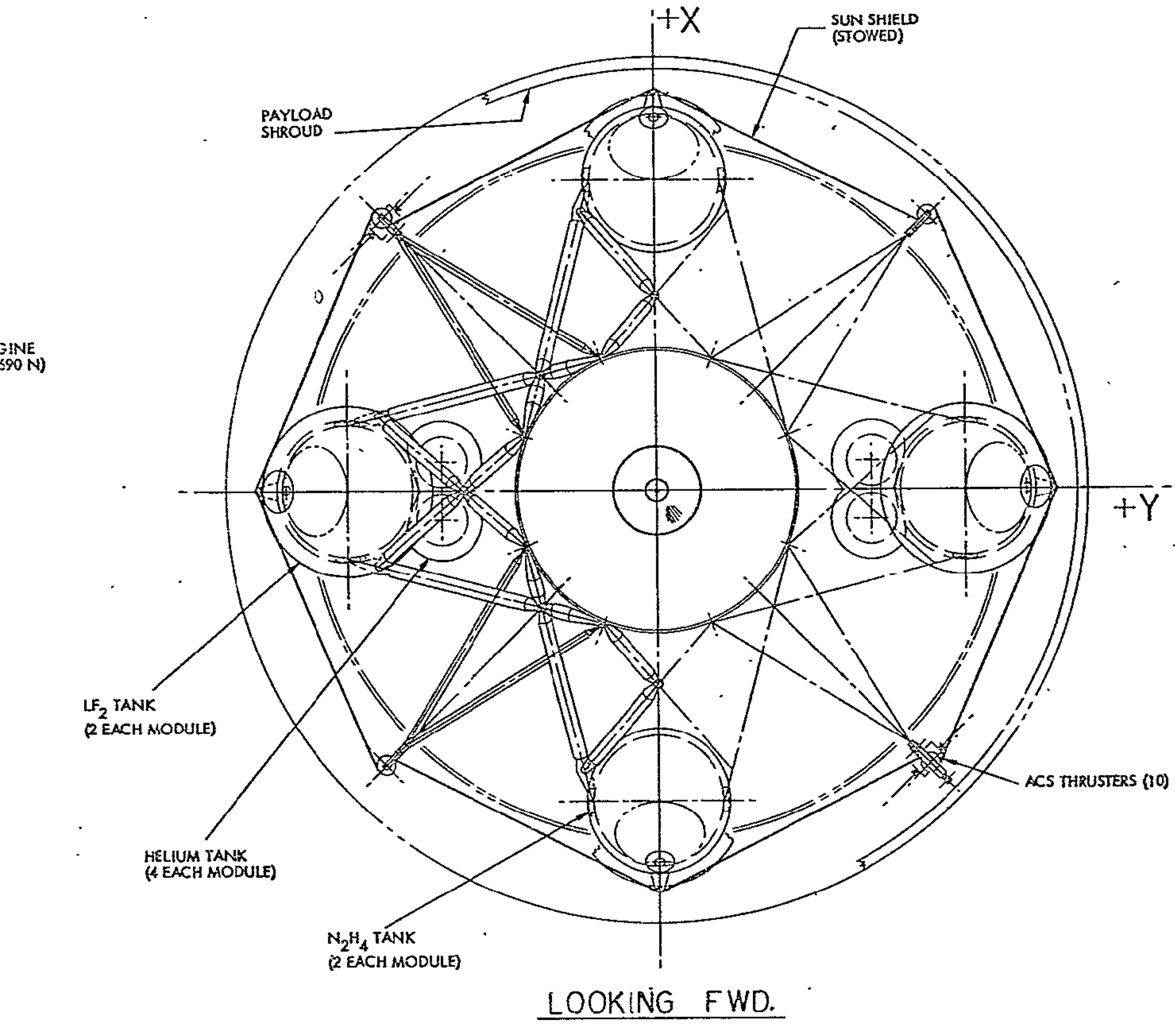


Figure 8. Propulsion Module Configuration for Pioneer Mercury Orbiter (Tandem Arrangement)

The spacecraft is shown, with the cylindrical sun shade in stowed configuration, mounted on a Centaur D-1S upper stage in the Shuttle cargo bay. This version of Module A is designed for space-storable propellants. The payload spacecraft is the Pioneer Venus orbiter with modifications required for the Mercury mission, e.g., a conical solar array adapted for thermal conditions at 0.31 AU (Mercury's perihelion) similar to the design used in the 1974 Helios spacecraft.

In the cruise mode the spacecraft's spin axis is maintained perpendicular to the plane of motion (and, hence, normal to the sun line), permitting the despun antenna to point continuously at earth with only small changes of elevation angle. For Mercury orbit insertion and other maneuvers the spin axis can be reoriented from the cruise attitude but must remain in an attitude normal to the sun line to assure continued protection of the propulsion module by the deployed cylindrical sun shade (see below).

The propulsion module contains four outriggered teardrop-shaped propellant tanks and four pressurant tanks, a central support cylinder and four support trusses that carry the propellant tanks. The 800-pound (3560 N) main engine mounted inside the cylinder is enclosed by a radiation shield. The support trusses are attached to the propellant tanks by mounting bosses located at the tank sides for efficient transfer of the axial load. The long support struts provide ample margin against a direct conductive heat transfer consistent with thermal separation requirements between the cryogenic LF_2 tanks and the adjacent warm N_2H_4 tanks.

The two tandem-mounted propulsion modules are connected by a V-band separation joint. A similar separation joint connects the lower propulsion module to the interstage adapter.

The N_2H_4 tanks are thermally insulated by multilayer insulation blankets. No insulation is used on the oxidizer tanks to permit radiation to cold space so as to maintain proper thermal balance at the desired cryogenic storage temperature. Four helium pressurant bottles are used which are attached in pairs and thermally coupled to the LF_2 tanks. This reduces total pressurant storage volume and tank weight. A 3-inch (7.6 cm) foam layer encloses the cold oxidizer tank and pressurant bottles to prevent frost from forming prior to launch.

The tanks are enclosed by secondary skins with a spacing of about 1 inch (2.54 cm) to:

a) Provide for propellant retention against leakage into the Shuttle cargo bay or at the launch site in the event of a leak*

b) Provide a cavity for chemical vapor detection to alert the Shuttle crew in the event of a leak*

c) Provide shielding against meteoroid impact.

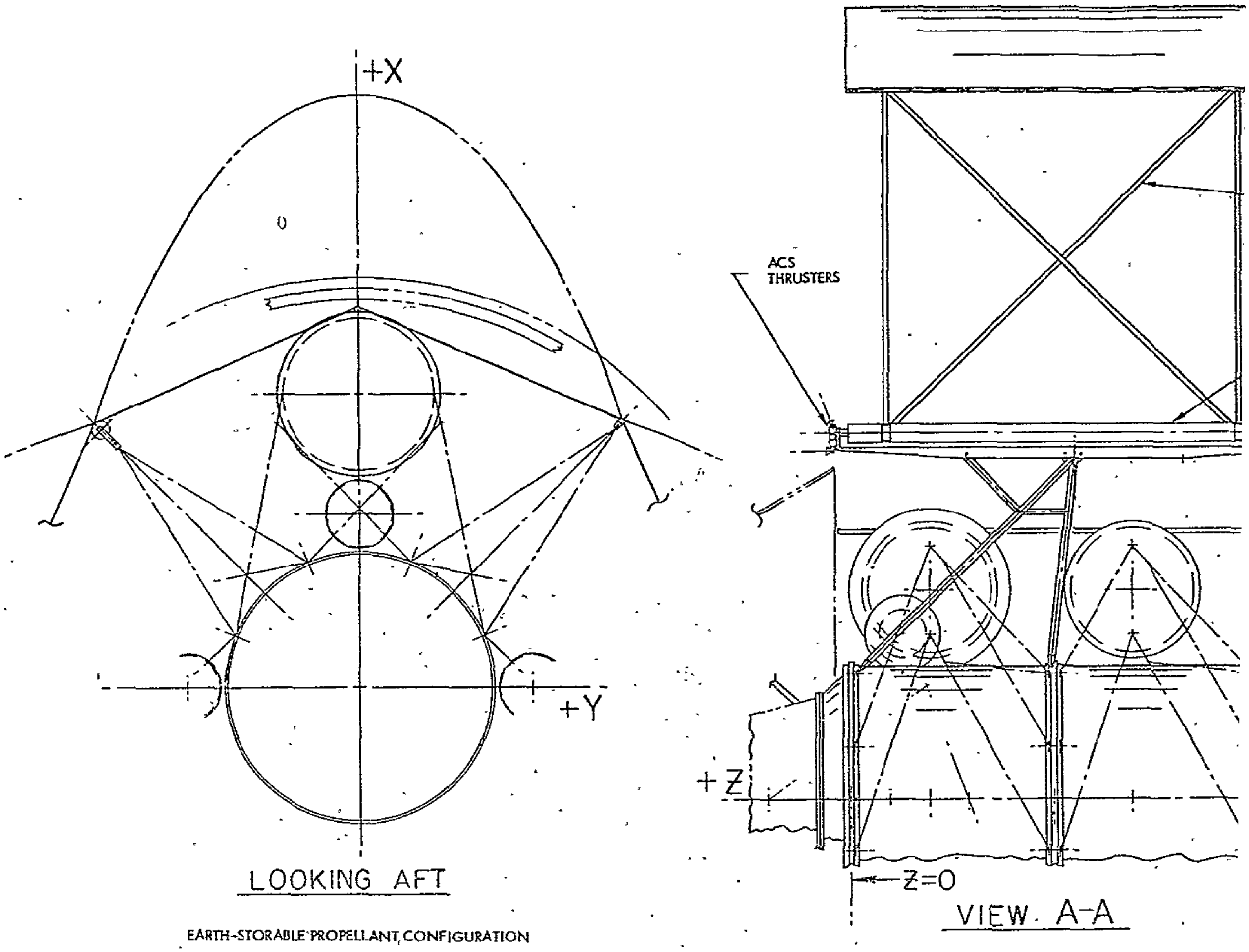
Another safety provision, not shown in the design drawing, is the addition of dump lines which permit rapid propellant disposal in the event of a leak, or in preparation of a Shuttle abort.

The spin-deployed sun shade (shown in Figure 9) protects the propulsion module against side-sun illumination and provides at least partial protection of the cold tanks against infrared Mercury radiation during passes over the dayside. It consists of a thin sheet of Beta cloth dispensed from four motor-driven roll-up mandrels. In the stowed configuration the sheet is tightly wrapped around the two propulsion modules, supported by the mandrels and propellant tanks. The deployment concept is illustrated in Figure 10. When fully deployed the sheet assumes a nearly circular cylindrical configuration retained in four places by the support arm and radially extended lanyards.

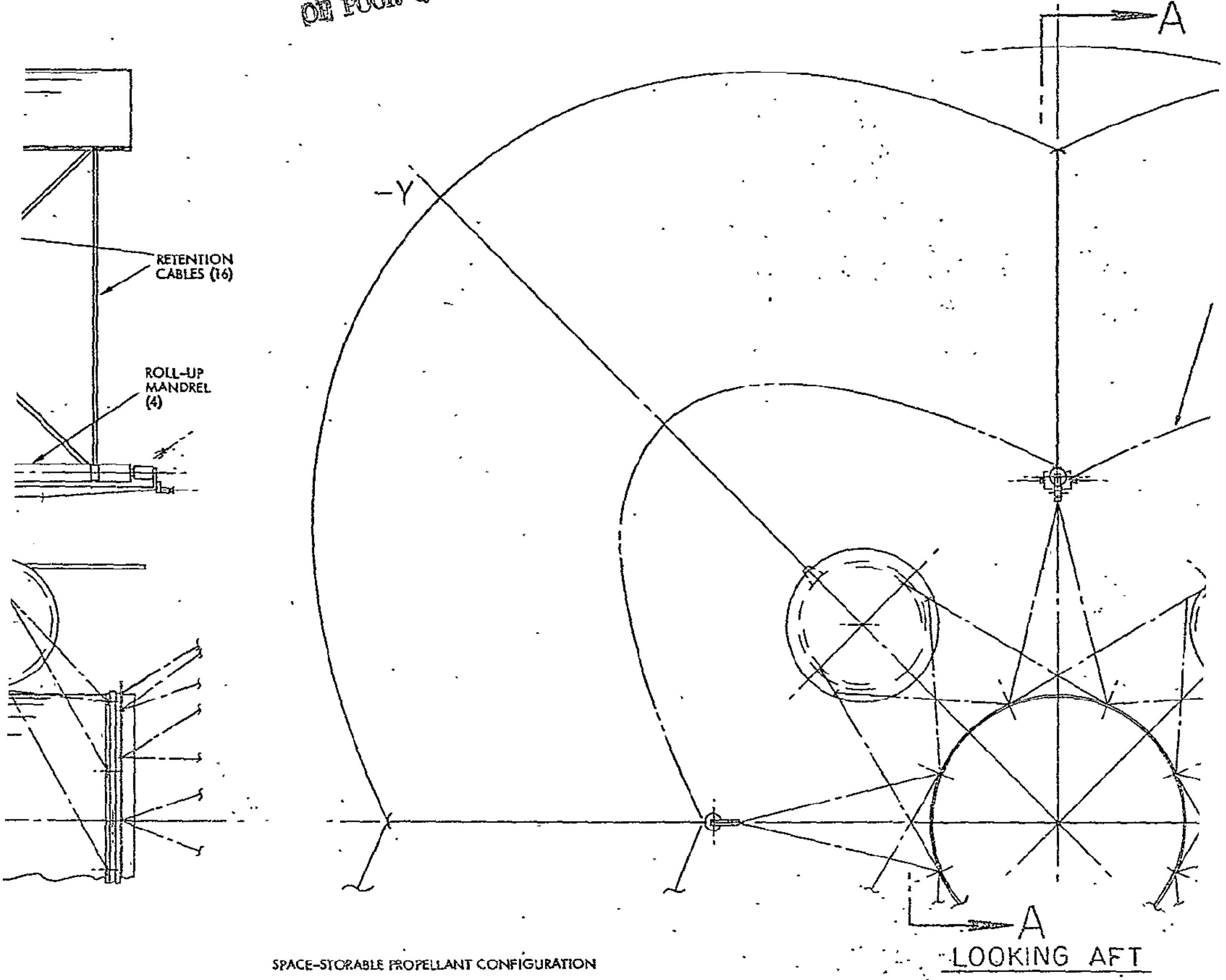
The large deployment diameter is necessary to give the fluorine tanks of the upper propulsion module a sufficient viewing factor of cold space to achieve a thermal balance at the upper limit of permissible cryogenic storage temperatures ($-250^{\circ}F$).

Before main thrust application the sun shade must be retracted since in the deployed position it cannot withstand large axial accelerations. After orbit insertion at Mercury, the shade can be redeployed to a smaller diameter since, with the lower propulsion module jettisoned, more unobstructed view of cold space in aft-direction is available to the upper module's cold tanks.

*In LF_2 tanks and possibly N_2O_4 tanks



EARTH-STORABLE PROPELLANT CONFIGURATION



SPACE-STORABLE PROPELLANT CONFIGURATION

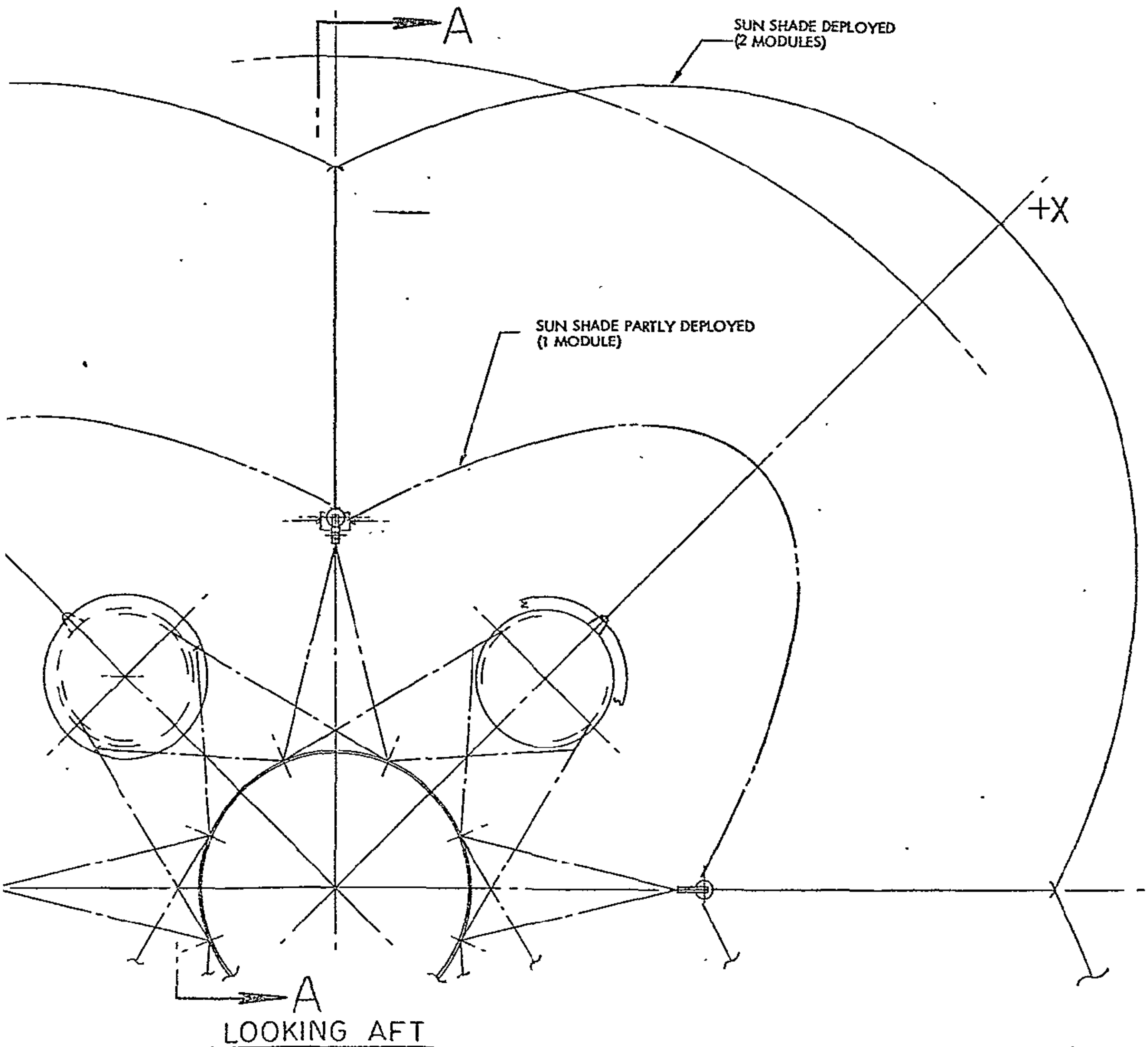


Figure 9. Pioneer Mercury Orbiter Deployed Sun Shade

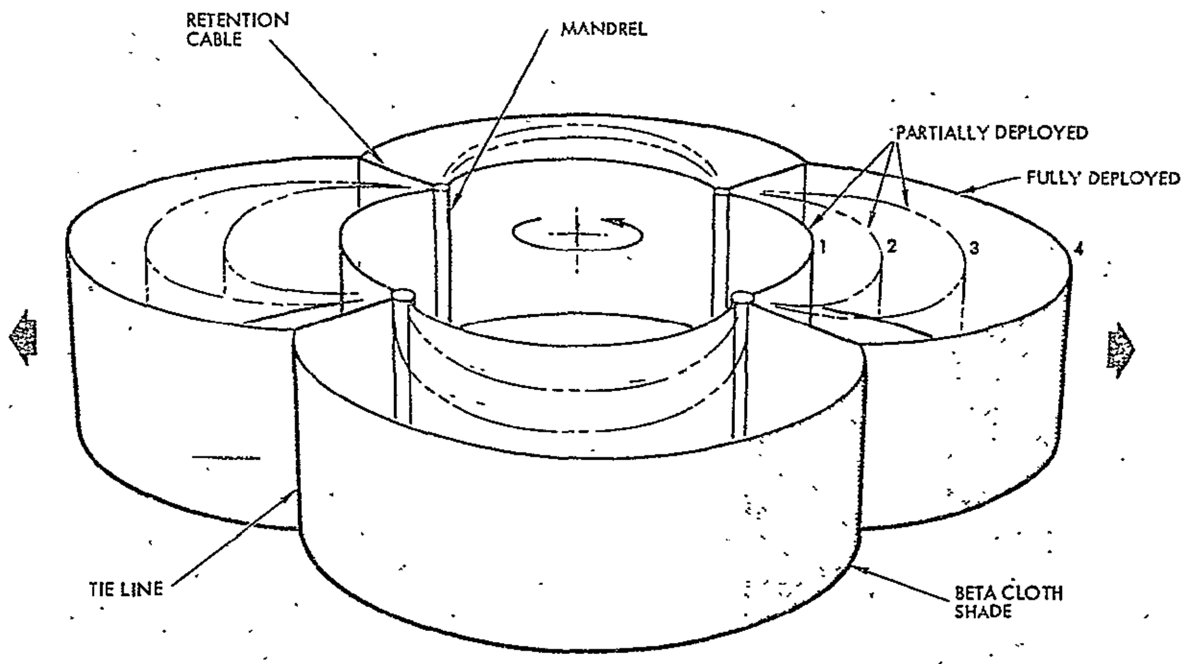


Figure 10. Deployment Sequence of Cylindrical Sun Shade.

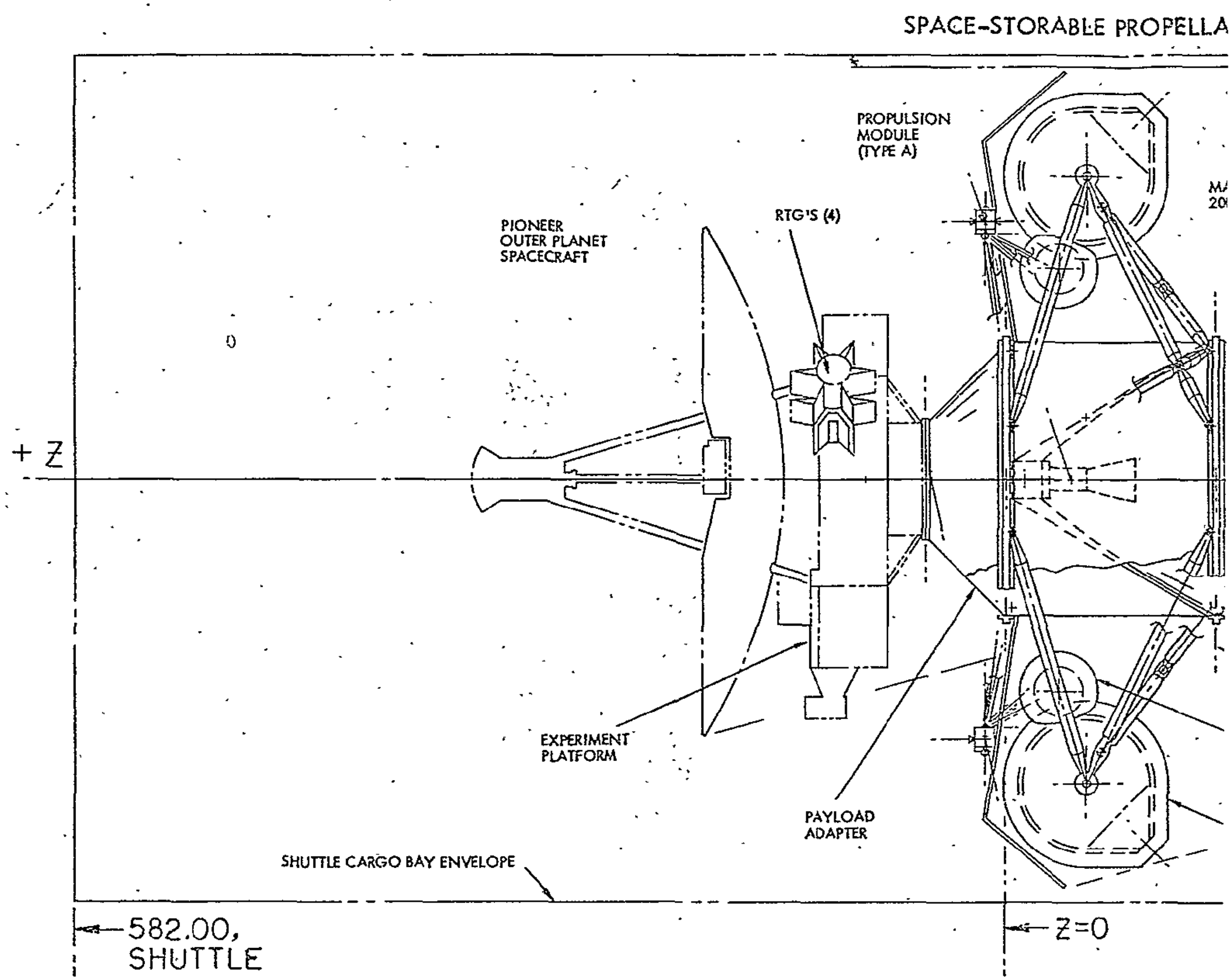
Low-thrust ΔV maneuvers and slow precession maneuvers can be executed without large deformation of the deployed sun shade. However, small shade deflections due to precession maneuvers will cause small transient nutations of the shade and center body. Damping due to propellant sloshing and shade deformations will cause these nutations to decay and restore steady-state alignment.

The earth-storable version of Module A requires a sun shade with much smaller deployment radius (see left drawing in Figure 9).

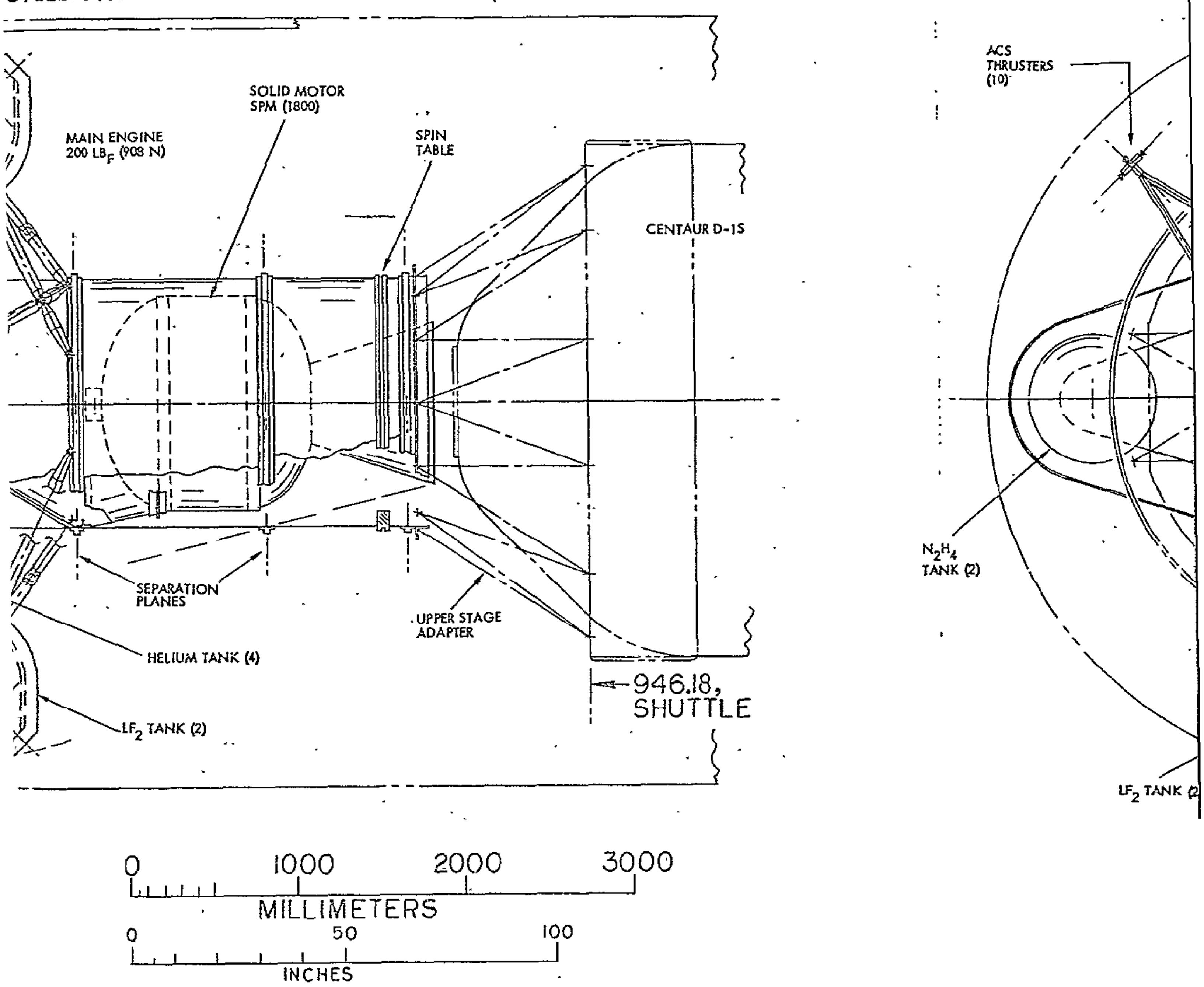
2.2.2 Outer-Planet Orbiters

Figure 11 shows Module A with space-storable propellants in the outer-planet orbiter configuration. The payload is a Pioneer 10/11 class spacecraft. The Shuttle/upper stage shown is the Centaur D-1S/SPM (1800) which has adequate performance for launching a Pioneer Saturn orbiter.

An interstage adapter truss supports the solid propellant kick motor and the flight spacecraft on the 10-foot (3.05-m) Centaur interface mounting ring. A spin table is provided to spin up the kick motor and payload prior to Centaur separation. For Uranus orbiter missions the


SPACE-STORABLE PROPELLA
PROPULSION
MODULE
(TYPE A)
PIONEER
OUTER PLANET
SPACECRAFT
RTG'S (4)
+ Z
EXPERIMENT
PLATFORM
PAYLOAD
ADAPTER
SHUTTLE CARGO BAY ENVELOPE
582.00,
SHUTTLE
Z=0

OPELLANTS



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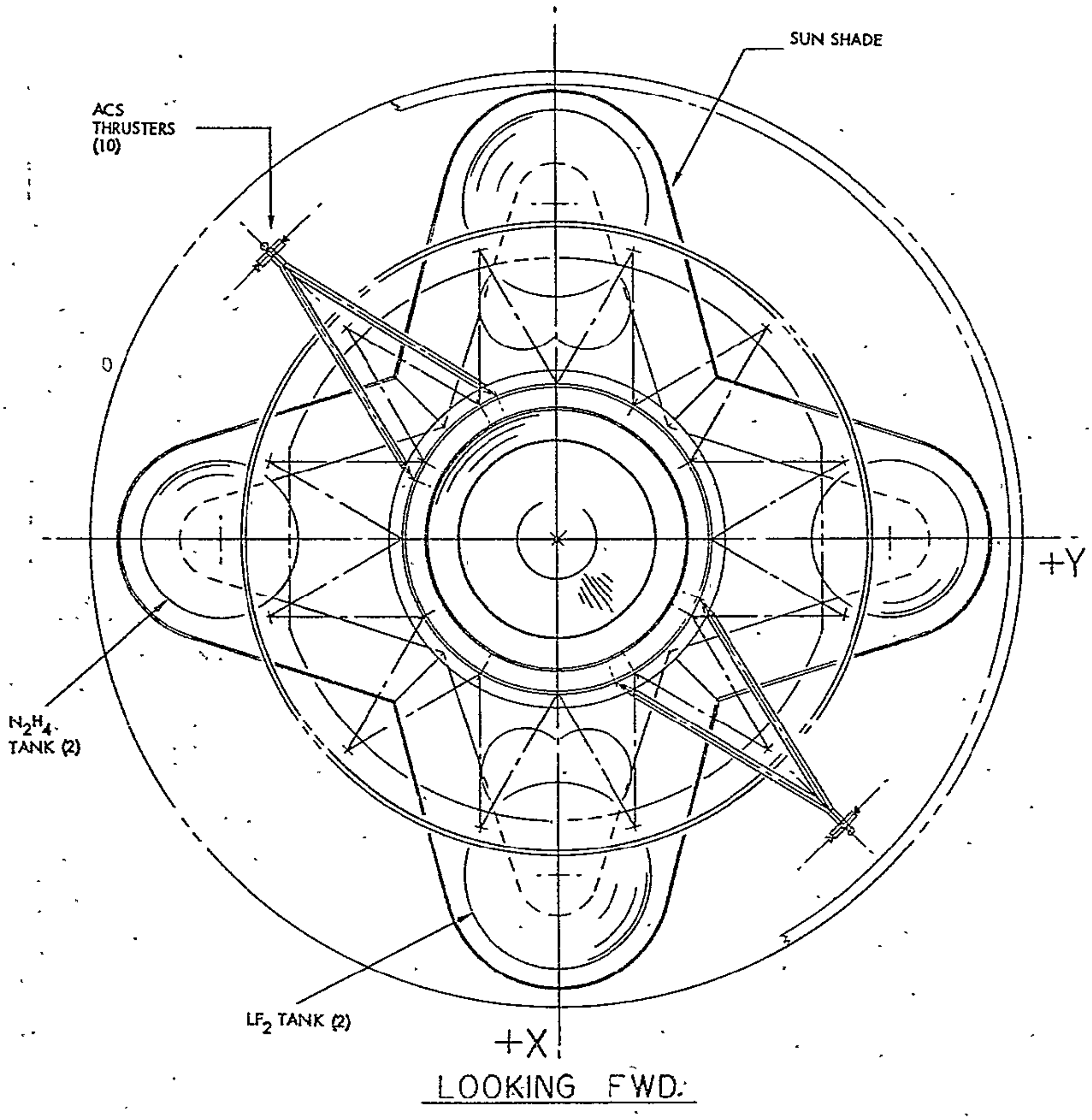

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Figure 11. Propulsion Module A Configuration for Pioneer Outer-Planet Orbiter

Space Tug/solid kick motor will be required as upper stage in place of the Centaur/kick stage. A different adapter truss is also required, in that case, to match the 14.5-foot (4.42-cm) Space Tug interface mounting flange.

The propulsion module is structurally identical to the tandem version used for the Mercury orbiter except as follows:

- Replacement of the 800-lb_f (3560-N) main engine by the smaller 200-lb_f (890-N) unit
- Addition of a four-leaf forward sun shade which extends beyond the high-gain antenna diameter to protect the propellant tanks against solar heating
- Removal of the cylindrical sun shade assembly used in the Mercury orbit configuration. Two of the support arms used in the Mercury orbiter are retained to mount auxiliary thruster assemblies
- The thruster assemblies are modified from the Mercury orbiter configuration.

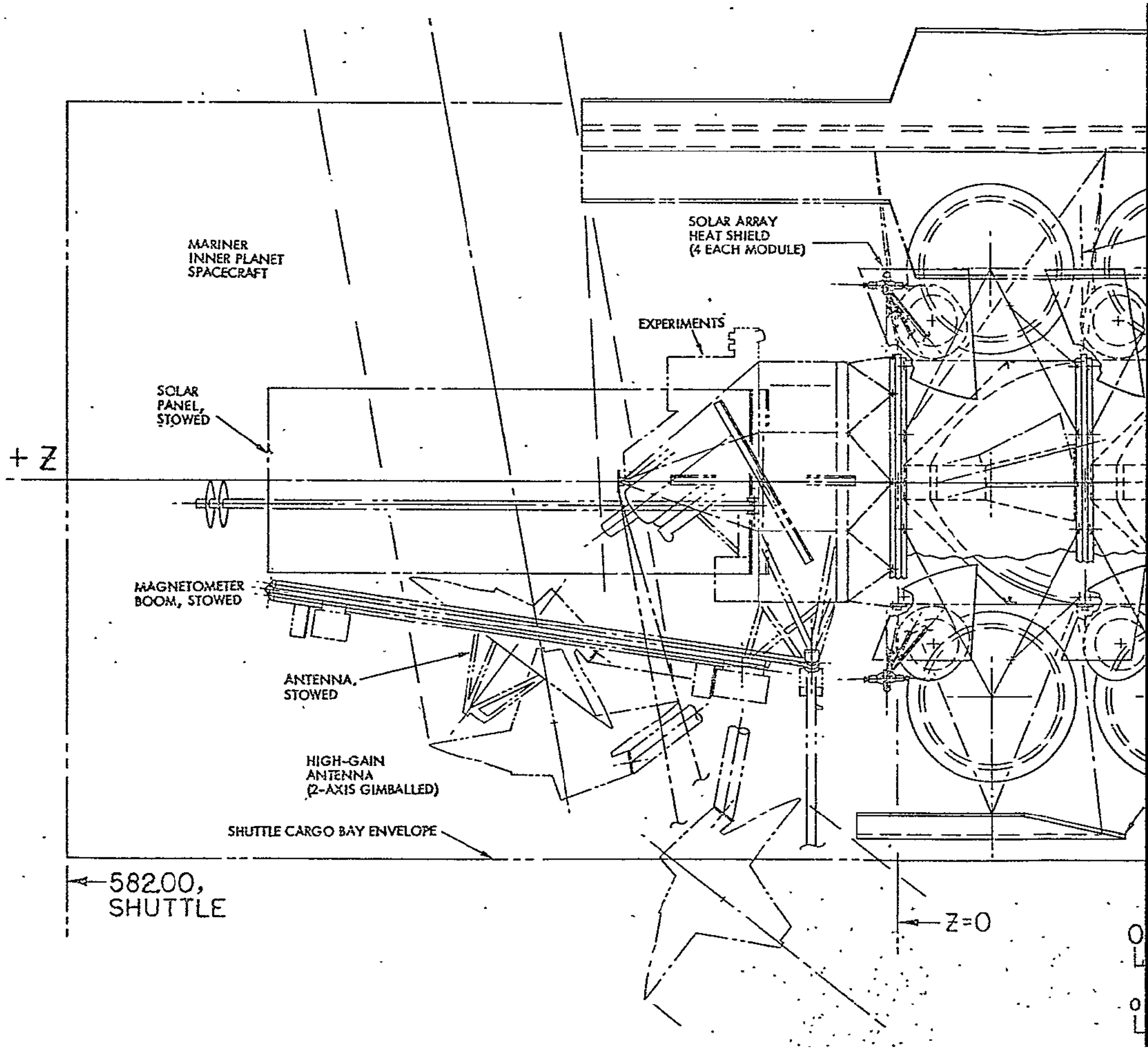
Prolonged side-sun illumination at angles greater than 15 degrees from the spin axis must be avoided with LF_2/N_2H_4 systems because of limited sun shade coverage. This implies that downlink communication via high-gain antenna must be interrupted twice for periods of several weeks during the early transfer phase. Communications coverage can be provided by the low- and medium-gain antennas during these periods.

In the earth-storable version of Module A, with propellant tanks adequately insulated against side-sun exposure, this constraint does not apply.

2.3 CONFIGURATION OF MODULE B (THREE-AXIS STABILIZED SPACECRAFT)

2.3.1 Mercury Orbiter

Figure 12 shows the selected propulsion module design for three-axis stabilized payloads, arranged in tandem for the Mercury orbit mission. The spacecraft is shown in stowed configuration mounted on the Space Tug in the Shuttle cargo bay. No solid kick motor is required for this mission.



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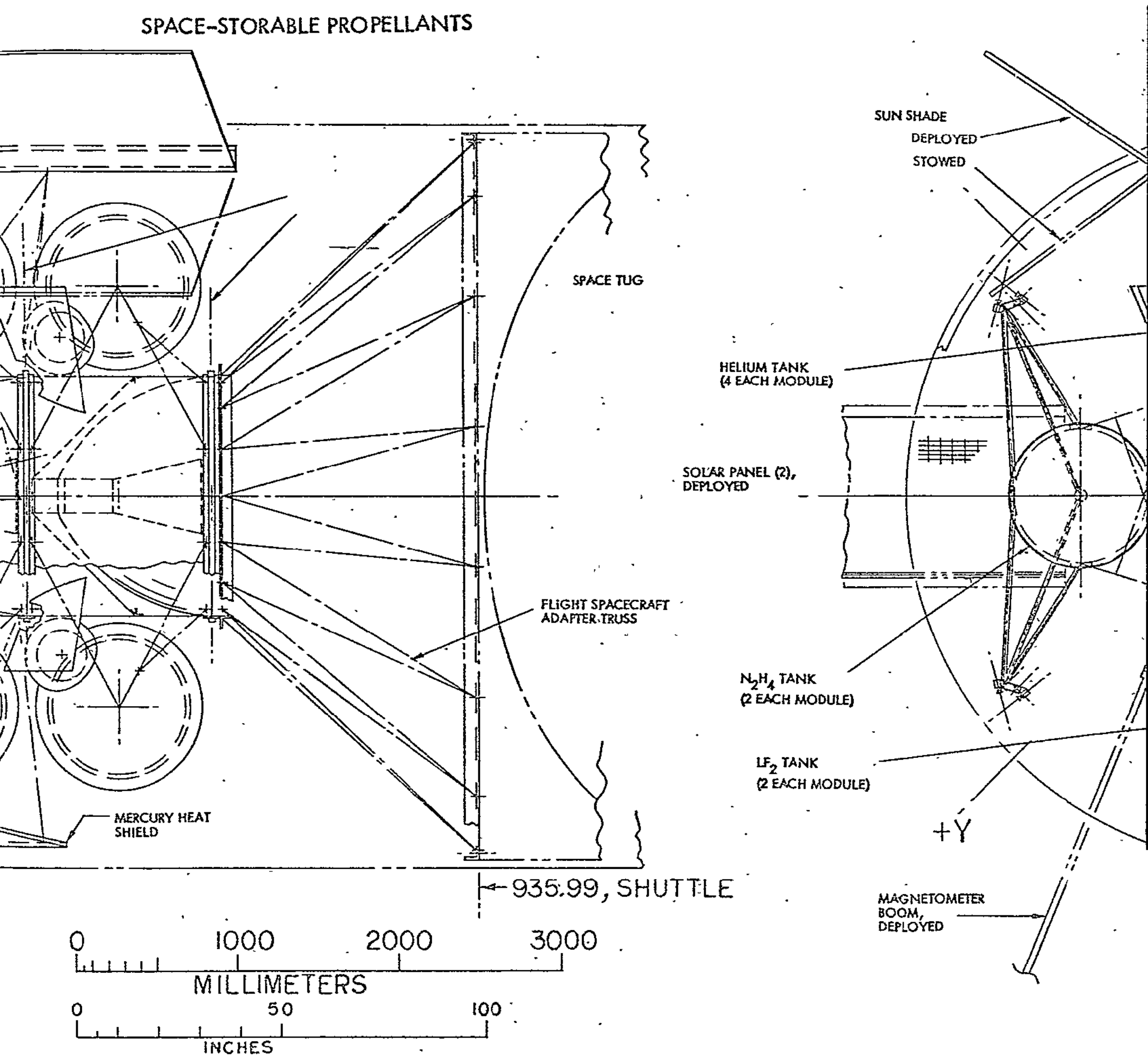
SPACE-STORABLE PROPELLANTS



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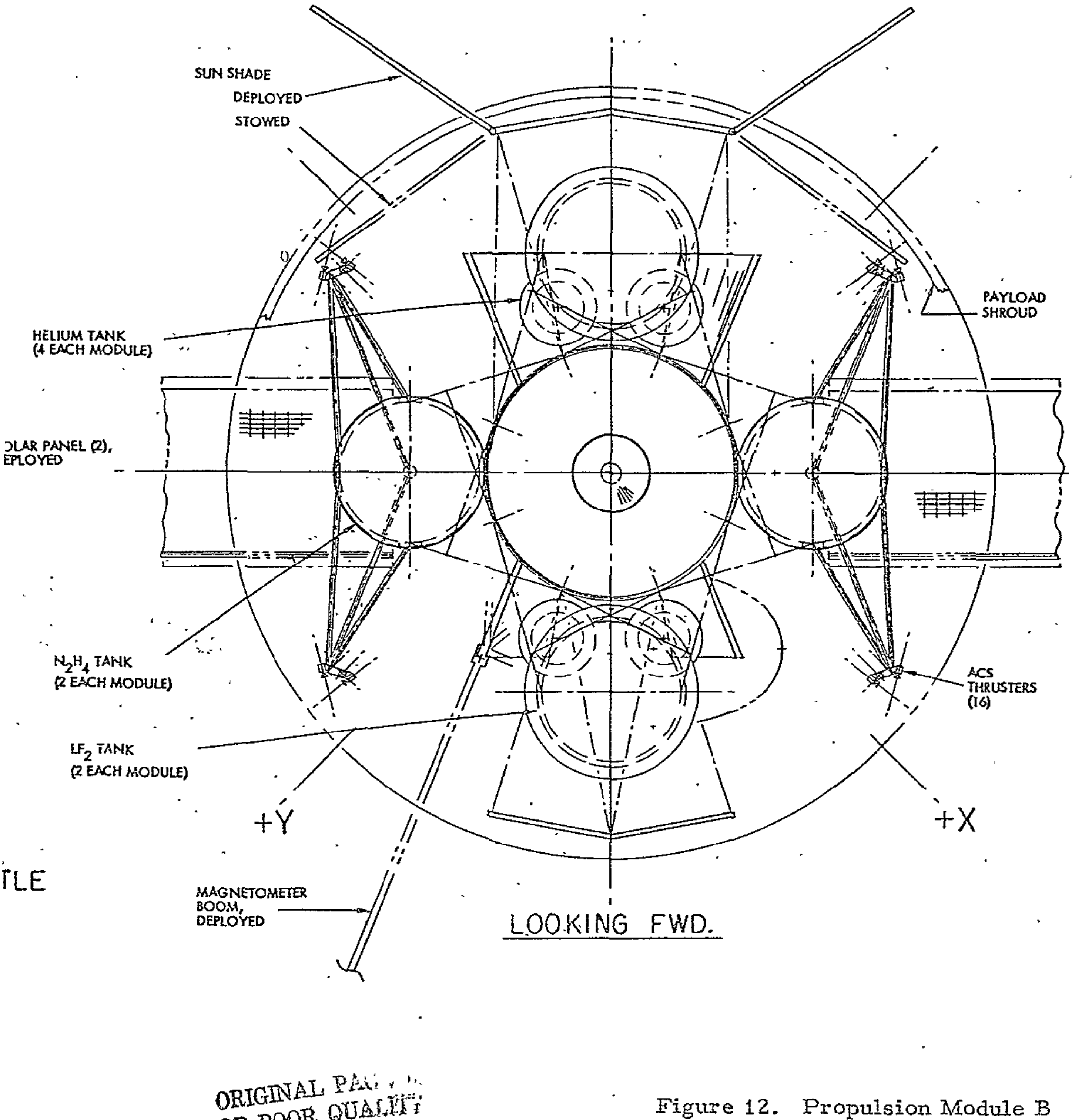

Figure 12. Propulsion Module B Configuration for Mariner Mercury Orbiter

The payload spacecraft is similar to Mariner 10, the Venus Mercury (MVM) flyby spacecraft, with modifications required to accommodate a different orientation relative to the sun. Thermal protection during the Mercury orbit insertion maneuver requires a side-sun shade. The side-sun orientation can be maintained during the cruise phase thus making the frontal sun shade used by MVM unnecessary.

As in the Mariner-Venus-Mercury spacecraft, solar panels are thermally protected against overheating with decreasing solar distances by gradual rotation from the initial sun-oriented attitude to a maximum tilt angle of 75 degrees. The required design modification only involves a rearrangement of the solar panel rotation joint. Guy wires are used to support the deployed solar panels against the maximum thrust acceleration of about 0.5 g.

The propulsion module configuration is similar to that of Module A using a hybrid support structure consisting of a central cylinder and four tank support trusses. Lateral mounting of the propellant tanks, while not required for mass distribution purposes as in Module A, facilitates the transfer of structural loads from vehicles above the propulsion module to those below. In the case of space-storable propellants it also facilitates thermal separation of warm and cold propellant tanks. The thermal design of propellant and pressurant tanks is similar to that used in Module A. The double-gimballed 800-lb_f (3560-N) main thrust engine is mounted inside the central cylinder, enclosed by a radiation shield.

V-band separation joints are used to connect the tandem-mounted propulsion modules to each other and to the launch vehicle adapter truss as in the design selected for Module A.

Propellant acquisition is effected passively either through capillary devices or by a propellant settling maneuver that uses auxiliary thrusters. Capillary devices are considered safe for the fuel (N_2H_4) tank but questionable for the oxidizer tank (LF_2) where corrosion products could cause clogging of downstream orifices in propellant filters and injectors. This passive method of propellant acquisition is more reliable in long-duration missions than the use of positive expulsion bladders and saves weight.

In the earth-storable propellant version of Module B capillary devices probably can be used in both the fuel and oxidizer tanks, making the propellant settling mode unnecessary.

The sun shade shown in the design drawing protects the propulsion module and the payload spacecraft in the cruise and maneuver attitudes. Prior to launch the hinged side panels of the sun shade are deflected inward to fit within the available cargo bay envelope.

Several other heat shields are required to protect the uninsulated cold tanks against heat radiated from the solar panels. Shielding against Mercury dayside heat flux is required only for the exposed fluorine tank of the upper module, since the lower module will be jettisoned at the time of Mercury orbit insertion. In the earth-storable version of Module B the auxiliary heat shields can be safely omitted.

2.3.2 Outer-Planet Orbiters

Figure 13 shows the outer planet-orbiter application of Module B with space-storable propellants. The payload is a Mariner Jupiter Saturn outer-planet spacecraft. The Shuttle/upper stage combination required for these missions is a Space Tug/SPM (1800).

The propulsion module is structurally identical to the tandem version used in the Mercury mission except for these modifications:

- Replacement of the 800-lb_f (3560-N) main engine by the smaller 200-lb_f (890-N) unit as necessitated by the limited load tolerance of the deployed appendages
- Omission of the large side-sun shade and heat shields that are required only in the Mercury mission
- Addition of small frontal sun shades to protect the fluorine tanks against direct sun illumination.

Spacecraft operation is constrained to avoid Z-axis orientations at angles more than 15 degrees from the sun line in the plane containing the fluorine tanks. As in the design of Module A, thermal control requirements dictate interruption of communication coverage via high-gain antenna during two periods early in the transfer phase when the earth-spacecraft-sun angle exceeds 15 degrees.

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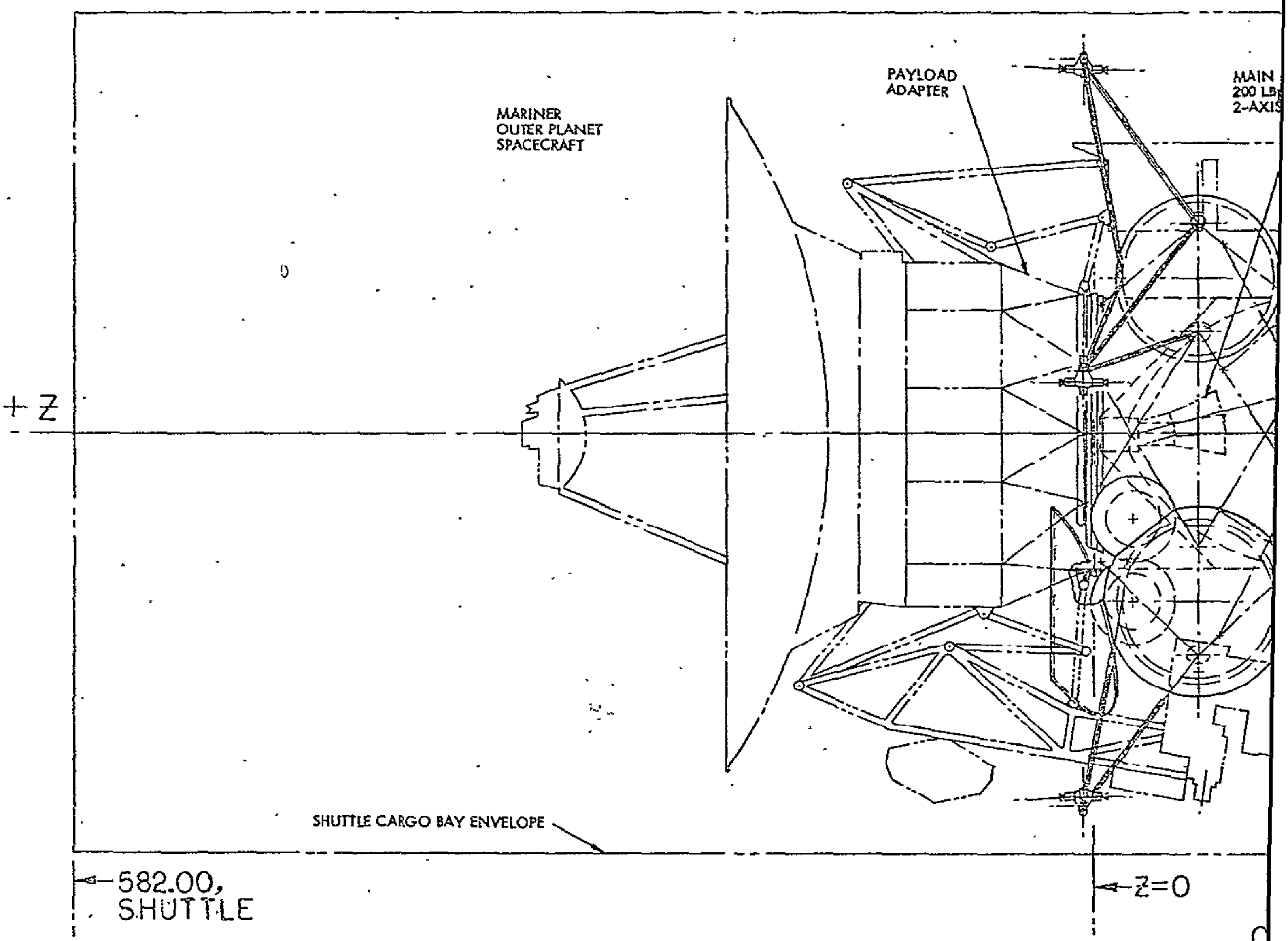

PAYLOAD
ADAPTER
MAIN
200 LB
2-AXIS
MARINER
OUTER PLANET
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SHUTTLE CARGO BAY ENVELOPE
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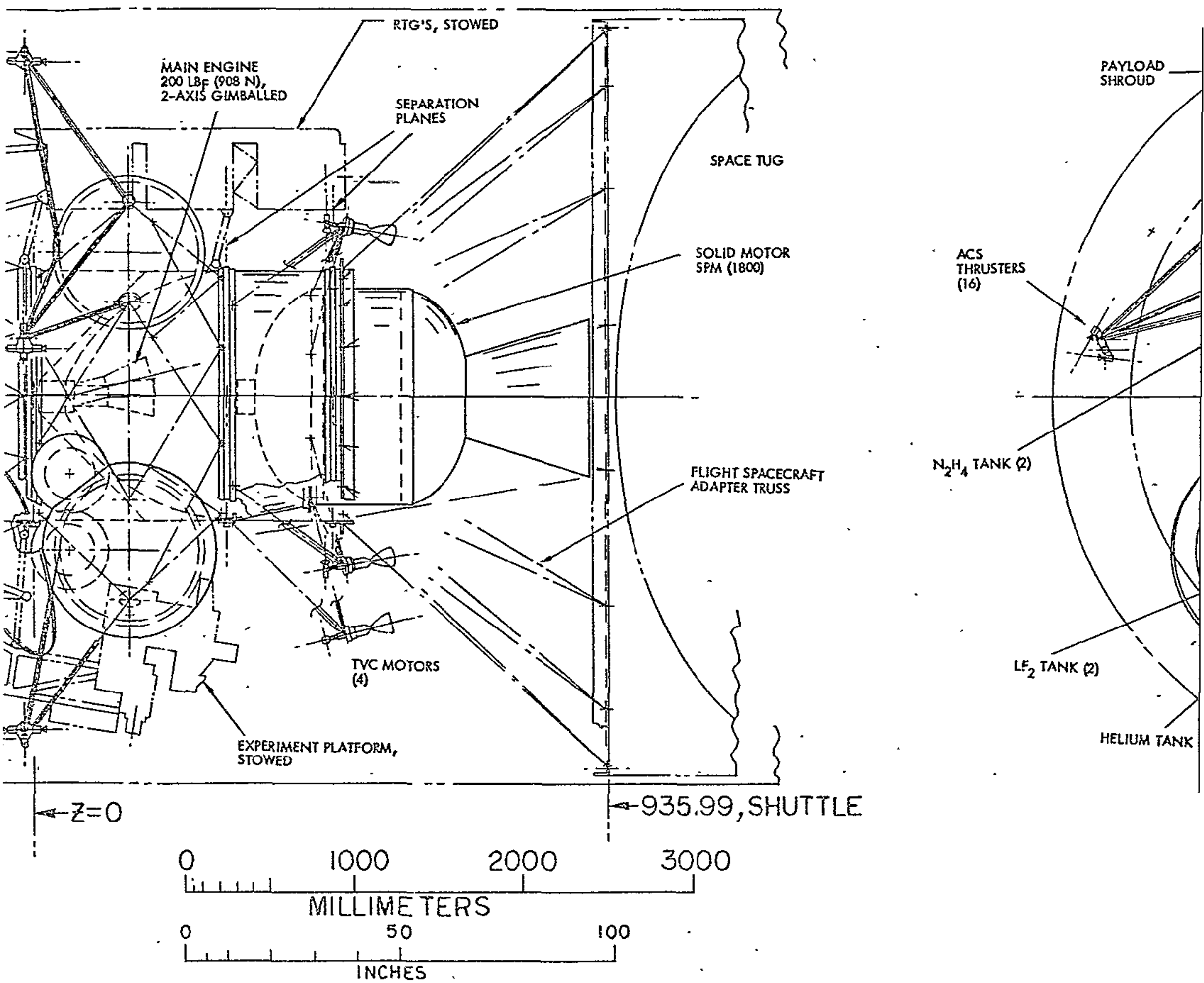
SPACE-STORABLE PROPELLANTS



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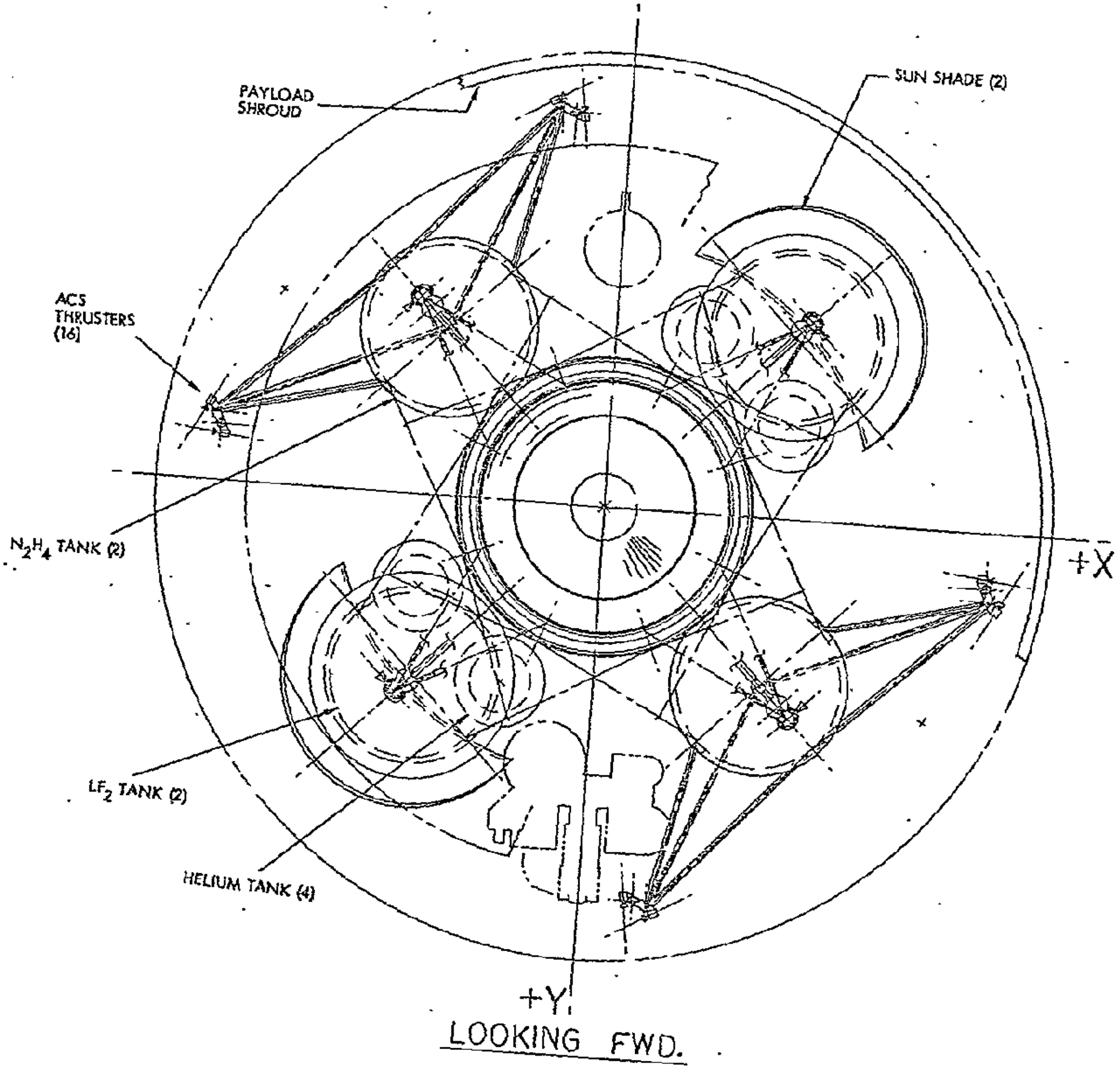

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Figure 13. Propulsion Module B Configuration for Mariner Outer-Planet Orbiter

2.4 OPERATING MODES

The configuration selected for Module A meets all orientation requirements of the Mercury orbit mission with regard to:

- Thrust pointing for effective orbit insertion
- Thrust pointing for secondary maneuvers
- Thermal protection
- High-gain antenna pointing
- Scientific instrument pointing.

During the transfer and planetary orbit phases the spacecraft will maintain a cruise orientation normal to the heliocentric plane of motion. This assures effective thermal protection by the sun shade and permits unobstructed earth pointing of the despun antenna with only small changes of elevation angle.

Off-nominal spacecraft orientations are acceptable provided that side-sun thermal protection and high-gain antenna coverage of earth are not lost as a result.

Thrust vector pointing options in the Mercury orbit insertion mode are related to the choice of approach trajectory for a given hyperbolic approach velocity vector, $\overline{V}_{\infty}$. Figure 14 shows a set of approach hyperbolas and periapsis location representative of the preferred mission opportunity in 1988. The aim angle, θ_{aim}, indicated in the B plane at left, determines the inclination of the approach orbit relative to Mercury's equator. Analysis of thrust pointing requirements for the various approach hyperbolas showed that an aim angle, θ_{aim}, of about 0 degree permits orbit insertion at lowest performance penalty under the constraint of side-sun orientation. Aim angles of 90 and 270 degrees corresponding to south polar and north polar approach trajectories would introduce a greater performance loss since the side-sun constraint requires a 12-degree out-of-plane thrust vector offset. Orbit insertion losses due to thrust vector offset are about 2 and 6 percent for the near-equatorial and polar orbit options, respectively.

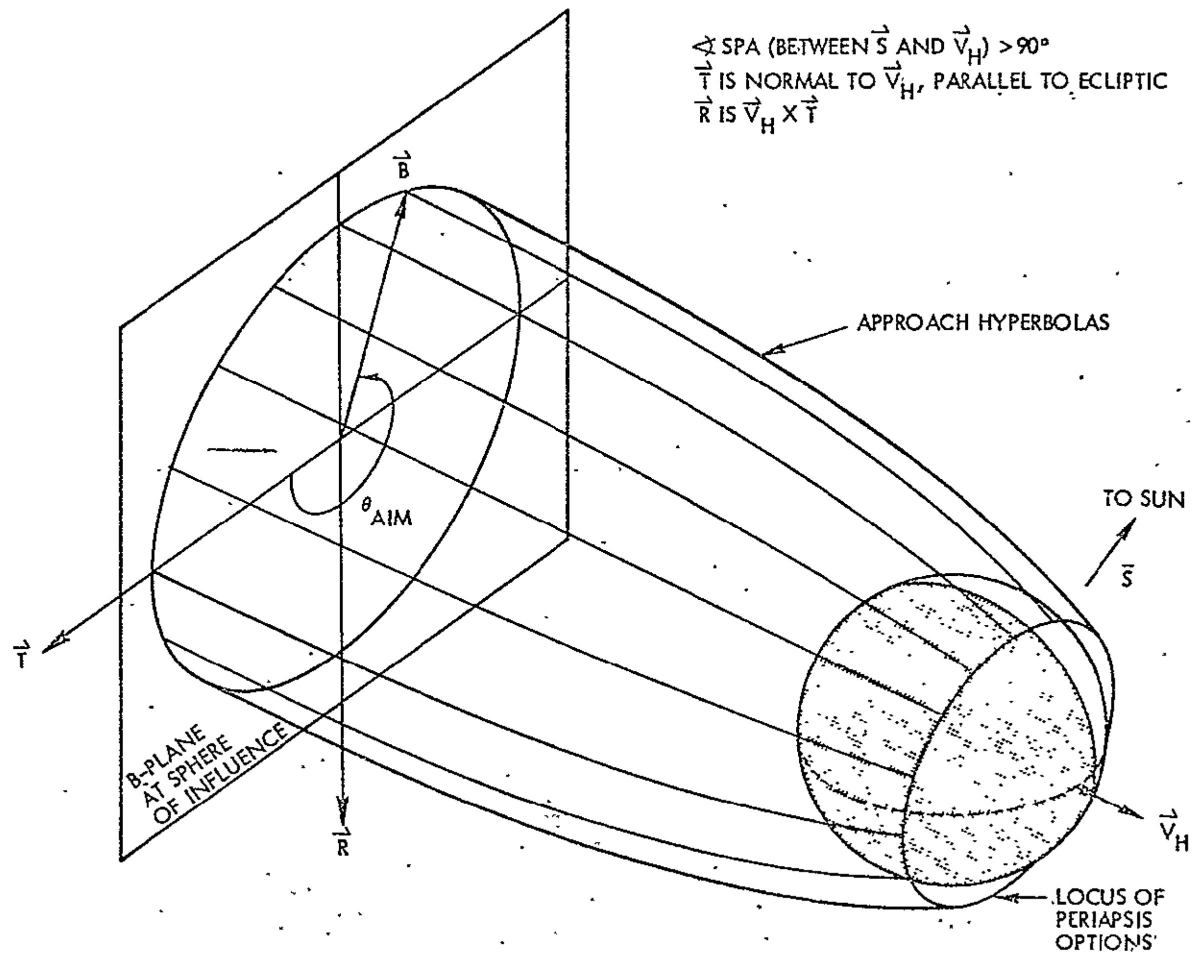


Figure 14. Mercury Approach Targeting Options

Mercury orbit injection for the three-axis stabilized spacecraft (Module B) is subject to fewer constraints. Insertion into a polar orbit (see Figure 15), scientifically more interesting than a low-inclination orbit, can be achieved without thrust vector offset losses. In fact, an optimum variable thrust pointing program using a gyro-controlled pitch rate maneuver can be more readily implemented than for the spin-stabilized spacecraft.

2.5 WEIGHT ESTIMATES

2.5.1 Propulsion-Module Inert Weights

Initial propulsion module size selection and inert weight estimates were based on empirical scaling relations. Since the propellant mass, the sizing of the module and, hence, its performance in the specified missions are very sensitive to inert weight, an iterative procedure was

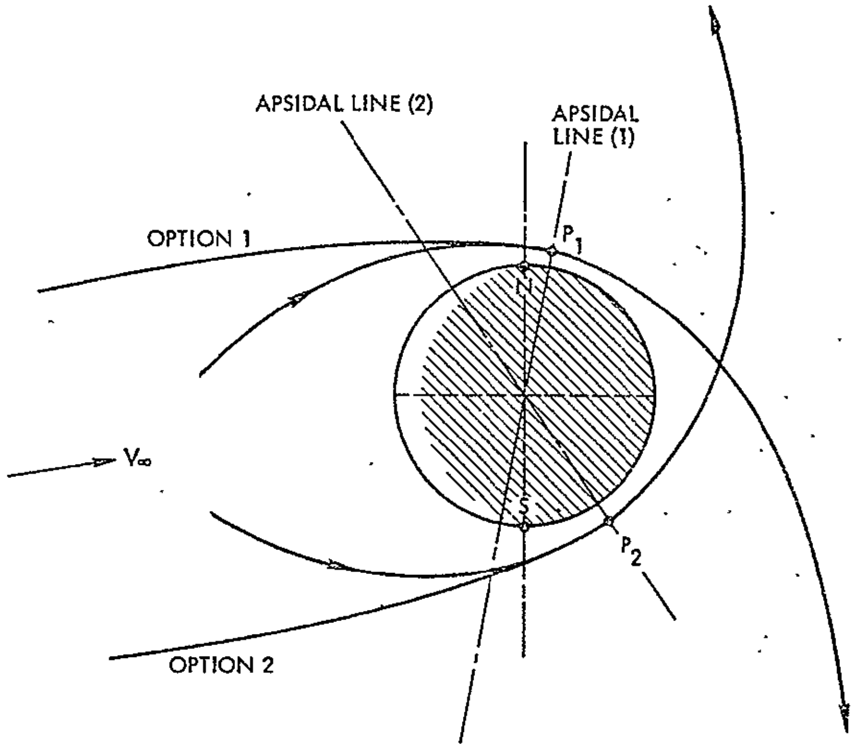


Figure 15. Two Arrival and Orbital Orientation Options for Mariner Mercury Orbiter

required to settle on the most appropriate stage size. Results of structural and weight analysis were used to update the initial weight estimates. Performance calculations were repeated on the basis of the improved weight data. The best estimate for inert weight variation with usable propellant mass (W_p) is given by

$$W_i = 0.1\ W_p + 120 \text{ kg.}$$

This relation differs from commonly used empirical scaling laws primarily because of the structural load conditions for which the multi-mission module is designed, primarily due to the tank mounting arrangement, the multi-mission/tandem configuration constraints, and Shuttle launch and abort load requirements:

- Only the support trusses and tank weights vary in proportion to propellant mass
- The weight of the central cylinder varies with the square root of the load since the structural design is based on critical buckling loads

- Other major structural components of the module are basically independent of propellant mass, being designed for crash load requirements of the Shuttle orbiter with propellant tanks empty.

The outriggered tank support concept was adopted to facilitate load path separation with two propulsion modules mounted in tandem, to facilitate thermal separation of warm and cold tanks in the space-storable propellant case, and to meet mass distribution constraints of the spin-stabilized Module A.

Extending these results to custom-designed propulsion modules leads to a similar relation

$$W_i' = 0.1\ W_p + 80\ \text{kg}$$

since the principal factors listed above imply a comparable dependence of structural weight on propellant mass.

2.5.2 Weight Summaries

A summary of weight estimates for propulsion Modules A and B are listed in Table 5 for both space-storable and earth-storable propulsion systems. These weight estimates result from structural analysis and propulsion system design and are in reasonably close agreement with inert weights used in the final performance iteration.

Table 5. Multi-Mission Propulsion Module Weight Summary in kg (lb_m)

Component	Module A				Module B			
	Earth-Storable		Space-Storable		Earth-Storable		Space-Storable	
Structure	77	(170)	59	(130)	91	(200)	68	(150)
Primary	60	(132)	47	(104)	73	(160)	54	(120)
Secondary	10	(22)	6	(14)	10	(21)	7	(16)
Uncertainty (10%)	7	(16)	5	(12)	9	(19)	6	(14)
Propulsion Subsystem	89	(197)	74	(163)	118	(260)	100	(220)
Propellant tanks[1]	39	(87)	29	(65)	52	(115)	39	(85)
Helium tanks + helium	20	(44)	11	(24)	23	(51)	15	(33)
Engine	12	(26)	16	(35)	12	(26)	16	(35)
Gimbal system	-	-	-	-	10	(22)	10	(22)
Propellant control system	5	(12)	5	(12)	5	(12)	5	(12)
Lines and fittings	5	(10)	5	(10)	5	(10)	5	(10)
Heaters and RHU's	1	(2)	1	(1)	1	(2)	1	(1)
ACS thruster assemblies	7	(16)	7	(16)	10	(22)	10	(22)
Thermal Insulation	5	(10)	5	(12)	4	(8)	4	(8)
Sun shade and Support[2, 3]	18	(39)	27	(60)	15	(33)	15	(33)
Contingency (6 percent)	11	(25)	10	(22)	14	(30)	14	(25)
Propulsion Module Weight (dry)	200	(441)	176	(387)	241	(531)	198	(436)
Usable Propellant	894	(1971)	551	(1215)	1272	(2804)	781	(1722)
Unused Propellant	9	(20)	5	(12)	13	(28)	8	(17)
Total Inert Weight	209	(461)	181	(399)	254	(559)	205	(453)
Total Module Weight (Wet)	1103	(2432)	732	(1614)	1525	(3363)	986	(2175)

(1) Includes 20 percent for secondary tank wall

(2) Weights stated are for Mercury orbiters. Outer-planet orbiters require about 12 kg less for this item.

(3) Heat pipe would lead to about 10 kg weight reduction in Module A (space-storable)

3. PROPULSION SYSTEM DESIGN

3.1 REQUIREMENTS

Propulsion systems to be used in the multi-mission propulsion module must satisfy criteria that are unique to the missions considered in this study, including the following:

- Mission life may approach 10 years
- Fluorine may be required as oxidizer to provide the high performance essential to the missions (high specific impulse)
- Multiple restarts are required with long dormant periods, e.g., major ΔV impulse at earth departure is followed by the planetary orbit insertion maneuver many years later
- The system must be compatible with different thermal conditions in extremely hot (Mercury orbiter) or cold (outer-planet orbiter) mission environments
- The system must conform with strict safety requirements of the Shuttle orbiter as launch platform, i.e., safety of propellant handling and storage; remote leak detection; rapid disposal of propellants by overboard dumping, etc.
- Multi-purpose use of propellants is desired, with main thrust and auxiliary thrust engines to be supplied by common tankage and pressurization system.

3.2 TECHNOLOGY STATUS

Earth-Storables

For systems using earth-storable propellants, a primary objective is extension of the demonstrated capability from about 2 years to about a decade. Propulsion systems using earth-storable bipropellants (N_2O_4/MMH) have demonstrated lifetimes on the order of 2 years in actual flight programs. Monopropellant hydrazine (N_2H_4) propulsion systems have a somewhat longer demonstrated life.

For earth-storable systems, the state of the art is represented by systems using cold-gas pressurized N_2O_4 and MMH with pressure-fed ablative, conduction or radiation cooled engines operating at 100 to 200 psi (7 to 14 bar) chamber pressures. Specific impulse performance

is 282 to 296 seconds. Spacecraft propulsion systems utilizing this propulsion technology include TRW's Multi-Mission Bipropellant Propulsion System (MMBPS); Mariner and Viking propulsion systems of the Jet Propulsion Laboratory (JPL); NASA's Apollo Service Module, Lunar Descent (LMDE) and lunar ascent propulsion systems; the Titan Transtage and several reaction control systems (RCS). The MMBPS, Mariner, and Viking are those most similar to the systems considered in this study.

Space-Storables

For space-storable systems with fluorine oxidizers the technology base is quite limited and a considerably greater advancement in the state of the art is necessary. Although technology efforts and advanced developments have been started, no fluorine system has been qualified or flown thus far.

JPL has successfully tested a complete (although not flight-weight) fluorine propulsion system at their facilities at Edwards Air Force Base, California, with good success (Reference 3). Specific impulse performance was approximately 363 seconds. In addition, they sponsored a TRW study under Contract NAS7-750, "Space-Storable Propellant Module Thermal Control Technology" (Reference 6), and have conducted several other related programs.

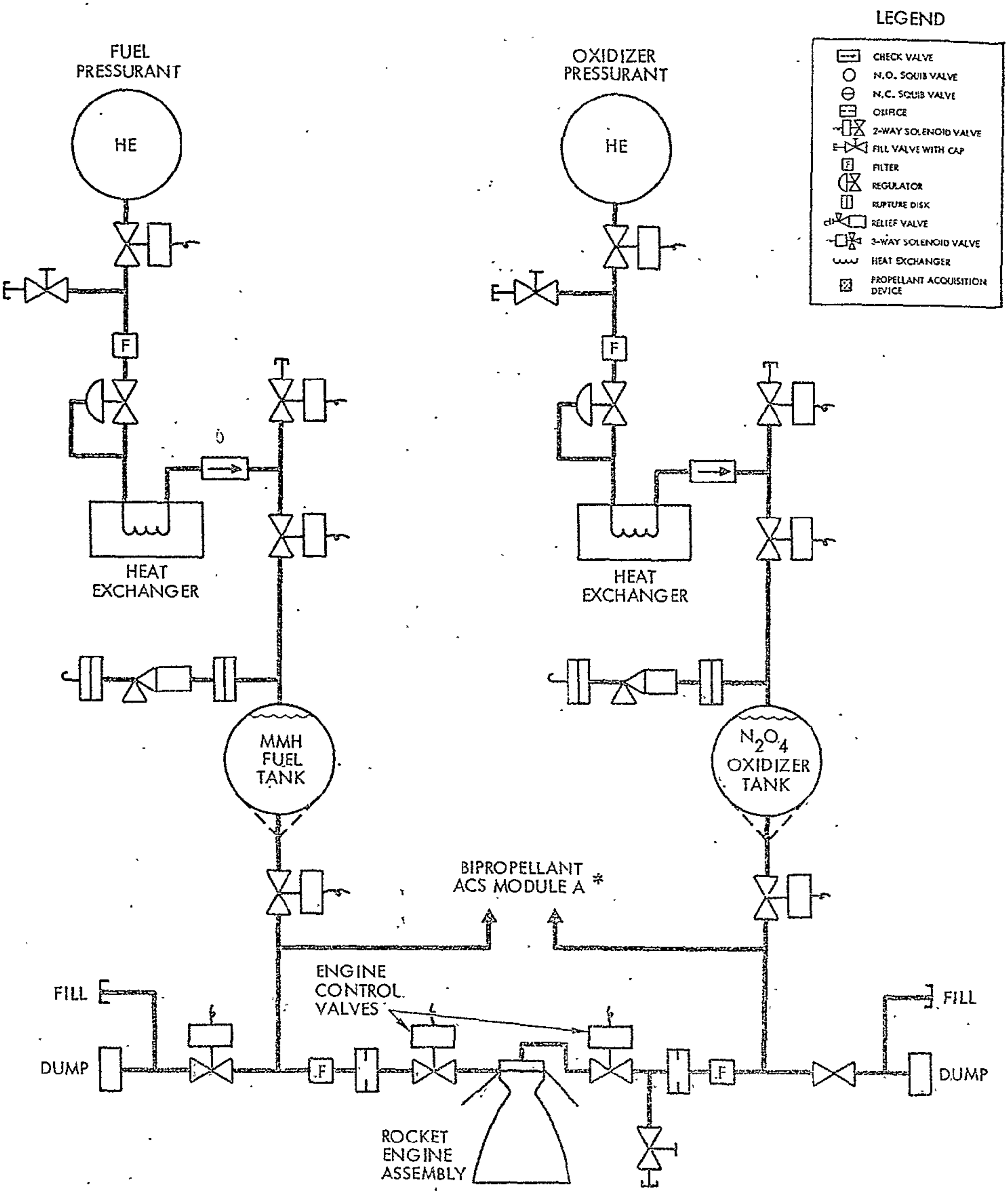
3.3 SYSTEM DESCRIPTION

A schematic diagram of the earth-storable propulsion system is shown in Figure 16A. This schematic reflects a) the experience described above, b) concurrent work by TRW on JPL Contract 954034, "Study of Safety Implications for Shuttle-Launched Spacecraft Using Fluorinated Oxidizers," (Reference 10), and c) results of the present study.

The LF_2/N_2H_4 schematic is derived from that in Contract NAS7-750 (Reference 6) which served as a state-of-the-art reference and point of departure (see Figure 16B).

The propulsion systems consist of separate fuel and oxidizer pressurization subsystems, tankage, and engine subsystems.

Each helium pressurization subsystem consists of tankage (1, 2, or 3 spherical bottles) made of Titanium 6Al-4V, isolation valve, fill valve,



MODULE B USES SPACECRAFT ACS

A. N_2O_4/MMH SYSTEM SCHEMATIC

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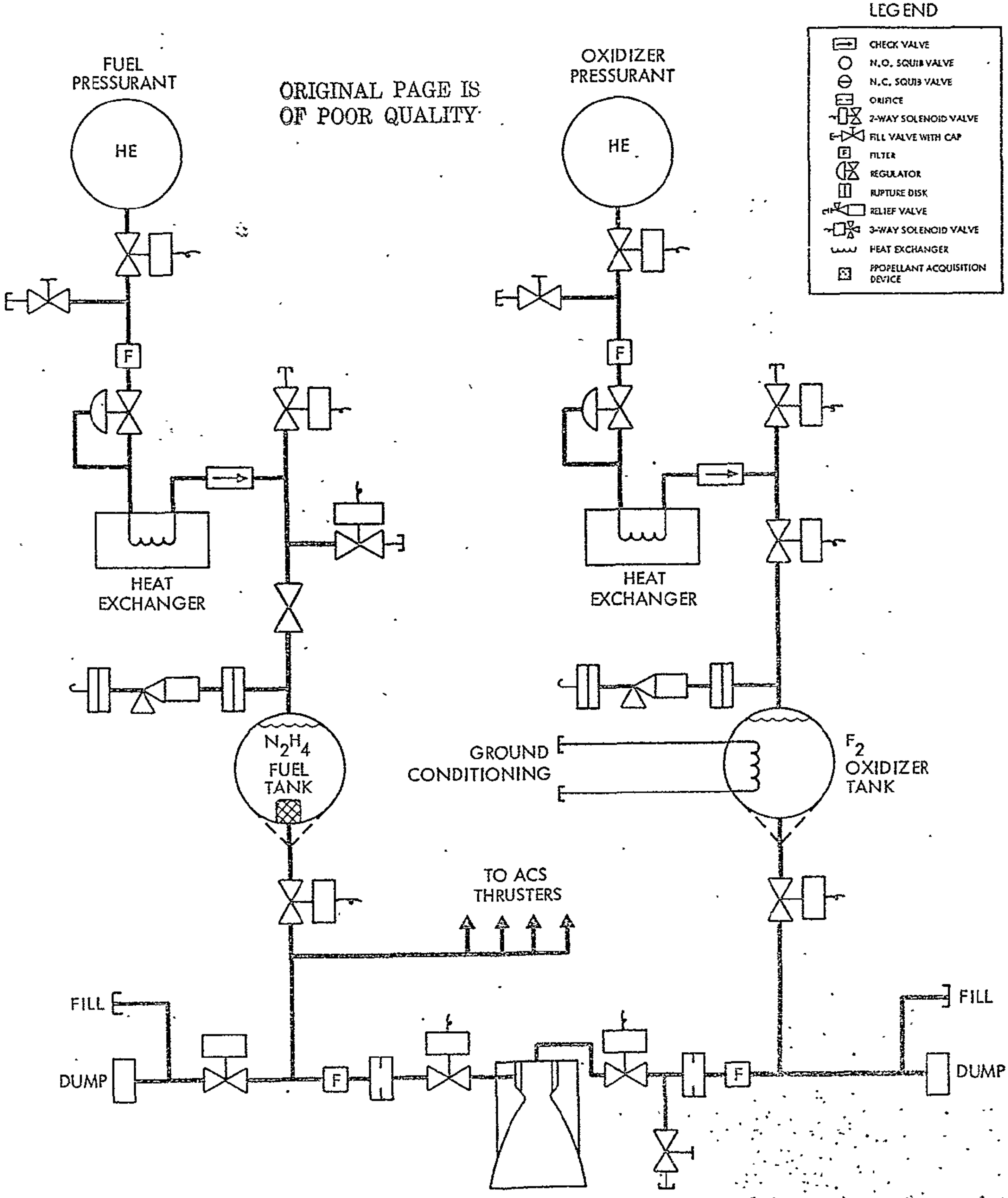


B. F_2/N_2H_4 SYSTEM SCHEMATIC

Figure 16. Schematic Diagrams of Earth-Storable and Space-Storable Propulsion Systems

filter, regulator valve, an optional heat exchanger, check valve, and servicing valve. The helium pressurization bottles are thermally connected to their corresponding propellant tankage except for the N_2H_4 pressurant, which is connected to the LF_2 tankage. A single, separable joint upstream of the tank isolation valve allows pressurization system disconnection.

The tank or propellant containment assembly in each case consists of four Titanium 6 Al-4V propellant tanks, an emergency relief valve with double redundant burst discs, isolation valves at the tank outlet and pressurization inlet ports, and remotely operated fill and dump valves.

A single, centrally located engine serves each module. Engine assemblies consist of two propellant filters; two orifices for calibrating mixture ratio; two engine control valves; and a thrust chamber assembly consisting of an injector, a combustion chamber and de Laval nozzle. Earth-storable propellant engines are radiation cooled; LF_2/N_2H_4 engines are ablative.

3.4 THERMAL OPERATING CONDITIONS

For best engine operation it is desirable to have the propellants at predictable temperatures so that flow rates are reproducible. The earth-storable propellants can be controlled by insulation and heaters to a comfortable range above freezing.

For the outbound missions and Mercury mission with Module B, the liquid fluorine can be maintained at a convenient temperature near its normal boiling point by control of heat inputs in balance with radiation to cold space.

Thermal analysis of the Pioneer class Mercury orbiter (tandem configuration, Module A) indicated that space storage of liquid fluorine must be accomplished at a higher temperature than for the outbound missions. With the selected configuration, using a deployed cylindrical sun shade of 15-foot radius, a fluorine storage temperature of approximately 117°K (-250°F) is about the lowest that can be achieved even with special coatings. At this temperature, the vapor pressure is 11.2 bar

(165 psia) and specific gravity of the liquid fluorine is only 1.25. This means that the engine design chamber pressure should always exceed the vapor pressure of approximately 165 psia. This leads to a desired engine combustion-chamber pressure of 200 psia or more. Preliminary chamber pressure optimizations not considering this effect also indicated a chamber pressure of 200 psia or higher. Because of state-of-the-art considerations, a design chamber pressure of 200 psia was selected.

Evaluation of the design concept for a spin-deployed cylindrical sun shade for Module A and consideration of development and test costs for this configuration have led to a search for possible alternatives that would use existing technology. The concept of a centrifugally actuated heat transporter for fluorine tank thermal control in the Mercury mission appears promising. However, this concept was introduced at a late stage in the study which allowed only a cursory examination of its characteristics. The technique is illustrated in Figure 17. The F_2 tanks are coupled to an aft-mounted radiator by nitrogen filled heat pipes. Because the spacecraft spin axis is normal to the solar vector this panel has a very low environmental heat input and can reject heat absorbed by the F_2 tanks at the low temperature required.

The heat pipes are attached to the outboard F_2 tank surface and configured such that the radiator plate is inboard. As a consequence, centrifugal action due to spacecraft spin motion aids in pumping the heat pipe working fluid, LN_2, from the radiator, where it is condensed, to the F_2 tank where it is evaporated. A wick and internal threading provide liquid control and assure that the working fluid wets the heat pipe wall. Heat pipe operating pressure at $-250^{\circ}F$ will be approximately 300 psia.

In the tandem arrangement the heat pipes for the upper-module LF_2 tanks must extend across the separation joint and to the radiator plate at the bottom of the lower module. These pipes are broken when the lower module is jettisoned. A second heat pipe and radiator plate (confined to the upper module) will be necessary to provide thermal control during the rest of the mission if the upper module is to be retained.

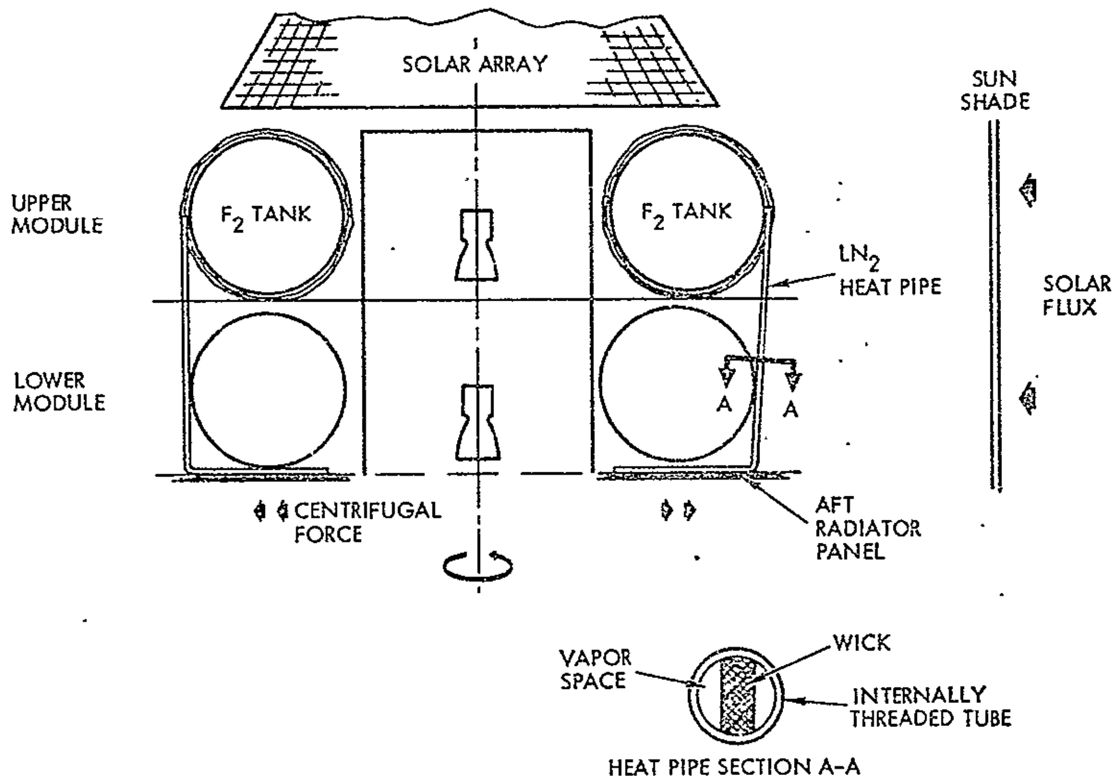


Figure 17. Heat Pipe Concept for LF_2 Cooling on Spinning Module

This thermal control concept has the following principal advantages:

- Replacement of large spin-deployed sun shade by a smaller (stationary) one. This saves weight and cost and reduces operational constraints
- Elimination of moving parts, hence greater reliability
- Lower development risk; easier, less costly verification tests
- Reduction of solar pressure unbalance
- Fluorine tanks can now be covered by thermal blankets and thus be given more protection against temporary heat inputs
- The system can be designed for lower oxidizer-temperature fluctuation and thus easier mixture-ratio control.

3.5 PROPULSION MODULE DESIGN FOR LONG LIFE RELIABILITY

3.5.1 Component Reliability

Mission durations range up to 10 years, while present systems have demonstrated operability for approximately 2 years. For 10-year durations, aspects usually not considered such as propellant tank life under pressure and/or corrosion, will become important. Long-term operating life, and storage life, as well as combination effects also increase reliability requirements on other components.

A reliable, long-life system design must take into account the best available information on environments the components will be exposed to. The planetary orbiter missions will be performed after the environments at the target planet will have been sampled by previous flyby missions and will be fairly well understood. One exception will be the Uranus mission: a 1985 projected launch date would precede the encounter date of a Mariner Jupiter Uranus mission launched in 1979 or 1980.

Tradeoffs performed during this study considered equipment redundancies, competing technologies, weight, cost, and practicality of implementation. Real-time life testing in a simulated environment of components intended for very long mission life generally is not feasible. Therefore, some overdesign and/or component redundancy is needed. Primary reliability concerns in the conceptual design phase include propellant acquisition, pressurant regulation, valve actuator implementation, propellant isolation and corrosion effects on tanks and propellant lines. Problem areas needing further research during the subsequent technology and hardware development phases were identified.

Propellant Storage and Acquisition

Corrosion is a principal concern in all conponents in contact with the oxidizers, especially fluorine. Impurities, such as water, can aggravate oxidizer corrosivity and lead to a slow pitting or crevice corrosion that may cause slow leaks in tanks or valves. The selected design approach is to minimize the number of components that are exposed to the oxidizer and to keep tank pressure low in order to reduce stress-corrosion. The main tanks can remain unpressurized initially until first use.

During operation, the tanks are pressurized with warmed pressurant. This permits tank pressure to relax after isolation from the regulated source during periods of inactivity and reduces the potential for stress-corrosion.

Passive surface tension propellant acquisition devices suitable for N_2O_4, MMH, and N_2H_4 tanks are currently being developed and perfected. As substitute for conventional expulsion bladders, they will avoid problems of leak, rupture, fuel and oxidizer corrosion, and degradation due to RTG radiation. They will thus provide much higher reliability in long-duration missions.

The state of the art in materials compatibility for LF_2 tanks is not highly developed. Capillary acquisition devices which could corrode and cause clogging of downstream filters, etc., will therefore be avoided with this oxidizer.

Valves

The chemical stability of ordnance material for squib-actuated isolation valves is another unknown for long space storage, especially in the RTG environment. Another concern is the power requirement for ordnance firing or, alternatively, the long-life integrity of wet slug tantalum capacitors as charge-storage devices for ordnance firing. Because of these questions, solenoid and motor-actuated valves are preferred alternatives.

Pressure Regulators

The relative merits of conventional pneumatic pressure regulators and mechanical pressure switches operating in a bang-bang mode with an on-off valve were considered. Pneumatic regulators were selected as a conservative approach since they are expected to have fewer and better known failure modes.

3.5.2 System Reliability

Tank Leakage

In the unlikely event of propellant tank penetration by a micrometeoroid, approximately one-fourth of the propellant of the module

will be lost. However, depending on the time of occurrence the mission may still achieve a partial success. The engine may have to operate at an off-nominal mixture ratio, at a performance loss.

To make use of the potential redundancy of the four-tank configuration selected, appropriate isolation of the two fuel tanks and two oxidizer tanks from each other is necessary. Otherwise, a leak occurring in one tank would cause the loss of the entire remaining fuel (or oxidizer) and pressurant. The isolation valves must be controlled automatically to prevent propellant in the undamaged tank from leaking out through the manifold line. Long communication time delays preclude timely remedial action by ground command in most cases.

Auxiliary Propulsion System Reliability

Premature wearout failures are possible, particularly in the ACS system, because of the number of operating cycles required of each thruster during long-duration missions, both in spinning and nonspinning spacecraft applications. The maximum number of limit cycles per control channel of Module B in Saturn and Uranus missions are estimated in excess of 2×10^5. Pulsed thrust operations by Module A thrusters, while generally lower, still will be of the order of 10^5 cycles. Sufficient ACS thruster redundancy is provided to reduce the effect of single-point wearout failures on mission success probability. With a total of 16 ACS thrusters in Module B and 10 in Module A, a sufficient number of backup modes are available to retain full attitude control and ΔV correction capability after a single thruster failure, and at least partial capability in most cases as a result of an additional thruster failure in any channel.

3.6 DESIGN CONSERVATISM

To achieve high system reliability a conservative approach was used in defining the propulsion system design, including the following:

1) Use of separate pressurant systems for fuel and oxidizer

2) Use of a safety factor of 2.0 for propellant tanks

3) Use of a pneumatic gas regulator

4) Avoidance of thin-gauge materials (e.g., capillary devices in oxidizer tanks)

5) Unpressurized storage of the propellants during inactive mission phases

6) Use of heated pressurant to permit automatic pressure decay in propellant tanks after pressurant shutoff

7) Redundant sealing of tanks after each propulsion event to prevent minor corrosive leakages

8) Capability for venting of the engine lines to prevent corrosion

9) Provision for degraded system operation in case of propellant loss due to a major leak (e.g., as a result of micrometeoroid penetration)

10) Inclusion of additional propellant reserve (10 percent) for contingencies.

3.7 SYSTEM SAFETY IMPLICATIONS

Results of a concurrent study performed by TRW on Shuttle safety implications (Reference 10) are directly applicable to this study and were used in assessing safety characteristics and providing safety features of the space-storable propulsion system. The following paragraphs give a brief summary of the objectives of that study and the results obtained.

The study objectives were:

1) To identify any unique propulsion system requirements resulting from the use of LF_2 as oxidizer in the propulsion system of a planetary spacecraft launched by the Shuttle orbiter

2) To compare the safety interfaces between the Shuttle (crew and hardware) and the spacecraft propulsion system when LF_2, instead of N_2O_4, is used as oxidizer.

Preliminary results of the study are discussed in the following paragraphs.

Technically, the problem for Space Shuttle-launched spacecraft consists of safely loading, transporting, and carrying into space a tank containing typically 1000 pounds of liquid fluorine which is a toxic, cryogenic, potentially corrosive fluid.

Feasibility of safe operation was investigated and the equipment and procedures necessary to maximize the chance of success determined.

Hazards are similar in kind, if not degree, to those encountered in use of nitrogen tetroxide (also a toxic oxidizer) in the Shuttle. It was concluded that residual risks from spacecraft using fluorine and nitrogen tetroxide oxidizers during ground and flight handling may be reduced by isolation of the oxidizer to only its tank. Operation of spacecraft propulsion in the vicinity of the Shuttle or launch site is not required. Proper recognition of the characteristics of both oxidizers must be given in spacecraft design and in ground and flight operations. This will require unprecedented safety precautions when used with the Space Shuttle.

Some of the key points are:

- Isolation of the oxidizers to only tanks with no oxidizer in piping while in transit.
- Design consistent with the best available practice, especially as to welding. An all-welded propellant containment assembly is recommended and double-wall construction is preferred.
- A development program which is conducted without unresolved technical difficulties, so as to provide assurance of safety.
- A safety development program instituted concurrently with the hardware development.
- Appropriate remote propellant loading facilities are provided and dedicated through siting. Leak detection and warning should be automated at the launch site.
- Appropriate processing and procedures instituted at the launch site and during flight.
- Appropriate staffing and training are implemented, including a propellant safety crew, from arrival of spacecraft on the pad until launch.
- Appropriate accommodations are provided in the orbiter, especially prevention of hazards from other systems. LN_2 cooling of LF_2 should be provided until liftoff, and propellant status instrumentation should be provided.
- A dump system is to be considered if external hazards to the LF_2 or N_2O_4 tanks from other systems in the cargo bay are possible.
- Suitable fluorine handling systems are provided for use at the Payload Changeout Facility (PCF).

- Fluorine resistant SCAPE suits are available and are utilized as required for protection of personnel.
- Copious quantities of water or other suitable damage limiting chemicals are available. Caution is required because under some conditions water could increase the damage due to a small fire.

The primary hazard to personnel was identified as propellant loading operations which are very similar in nature to routine transfers from the truck trailers used during delivery of fluorine to industrial users. These operations should be accomplished in an area reasonably remote from personnel and facilities concentrations.

Other important potential hazards are related to the transportation and installation of the loaded propulsion system, where great care must be exercised.

Residual hazards during flight in the Shuttle cargo bay from a propulsion system which has been loaded, stored, transported and installed appear low, provided that hazards are minimized to the propulsion system from other systems also in the cargo bay.

4. PERFORMANCE

4.1 LAUNCH VEHICLE CHARACTERISTICS

Performance characteristics of a variety of Shuttle upper stages, were computed based on best available (projected) mass data and I_{sp} values, and including realistic performance penalties and losses ("gravity loss") during the launch phase. Upper stages specified by NASA/Ames as candidate launch vehicles for the missions considered, include Centaur D-1S, Dual Transtage and Space Tug and several solid-propellant kick motors. (See Volume II for curves of payload versus injection energy C_3.)

4.2 MERCURY ORBITER PERFORMANCE

The multi-mission propulsion module was sized to meet Mercury orbiter requirements, assuming two modules operating in tandem. The required propellant mass and the propulsion module inert weight were obtained from the performance iterations previously discussed. Table 6 lists the resulting Module A and Module B mass characteristics for earth-storable and space-storable propellants. It also presents the injected gross spacecraft mass and indicates which launch vehicles are adequate to perform the mission.

The performance analysis of Module B used an optimum, variable thrust vector pointing program for Mercury orbit insertion and determined the optimum time of thrust initiation. The performance analysis for Module A assumed a fixed thrust vector orientation, nearly tangential to the flight path at periapsis but with a small in-plane thrust vector offset to meet the side-sun protection constraint.

The preferred orbital orientation in the case of Mariner missions is near polar with an approach hyperbola arriving over the north or south pole. This mission profile has the advantage of high latitude coverage and fuller exploration of the physical environment, i.e., the magnetosphere, of Mercury.

Table 6. Mass Characteristics of Mercury Orbiters — Module A and B

Flight Vehicle, Propellant Type	Weight, kg (lb_m)			Candidate Shuttle Upper Stages (No kick stage required)	Injected Weight Capability kg (lb_m)
	Propellant Weight[1]	Inert Weight[1]	Gross Weight		
Pioneer (340 kg)/Tandem Module A[2]					
Earth-Storable	894 (1971)	209.4 (462)	2546 (5614)	Dual Transtage	3900 (8600)
				or Centaur D-1S	5250 (11,600)
Space-Storable	551 (1215)	175.1 (386)	1792 (3951)		
				or Titan 3E/Centaur D-1T (for reference only)	3300 (7277)
Mariner (550 kg)/Tandem Module B[3]					
Earth-Storable	1272 (2805)	247.2 (545)	3588 (7912)	Dual Transtage	4000 (8820)
				or Centaur D-1S	5300 (11,700)
Space-Storable	781 (1722)	198.1 (437)	2508 (5530)		

(1) Each module
(2) Module A uses fixed thrust orientation, 5 degrees offset from optimum (near-equatorial orbit)
(3) Module B uses variable thrust pointing program (near-polar orbit)

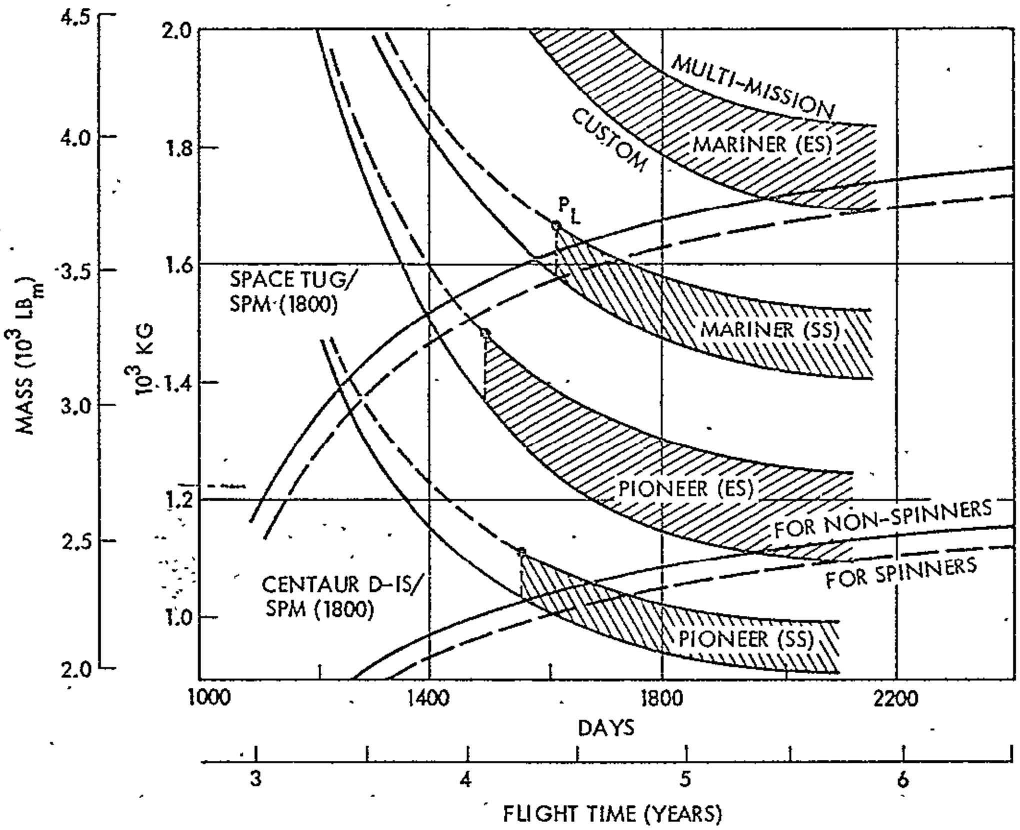
The reference mission adopted for the Mercury orbiter is one of two specified favorable launch opportunities in 1988 (see Reference 12). Launched on 12 March 1988 with a C_3 of 25.8 km_2/sec_2 and arriving at Mercury on 26 March 1990 the spacecraft performs two successive Venus swingby maneuvers, one of which requires a major ΔV expenditure ($\approx$200 m/sec). Compared with the launch opportunity in June 1988 a total (ideal) maneuver velocity reduction of 350 m/sec and a correspondingly large propellant saving is possible when using the earlier launch date. The mission with the lowest maneuver requirement was adopted in the interest of minimizing the multi-mission module size and inert mass, and reducing outer-planet mission performance penalties.

4.3 OUTER-PLANET ORBITER PERFORMANCE

4.3.1 Saturn Orbiter

Figure 18 shows the performance of earth-storable and space-storable propulsion systems in the Saturn orbit mission in a plot of injected weight requirements versus flight time. Launch vehicle performance curves represent the Centaur D-1S/SPM (1800) and Space Tug/SPM (1800). Intersections of the spacecraft weight requirements curves (sloping down) and the launch vehicle capability curves (sloping up) indicate the minimum flight time. Four performance bands are shown in the plot to represent the characteristics of Pioneer and Mariner class orbiters and space-storable and earth-storable propellants. Conservative payload weight estimates of 408 kg (900 lb_m) for Pioneer class, and 680 kg (1500 lb_m) for Mariner class outer-planet orbiters have been assumed in the calculations yielding the performance curves shown in Figure 18. The effect of parametric variations of payload spacecraft weight will be discussed below. Performance of the multi-mission stage is indicated by the upper boundary curve of each shaded band of injected weight requirements, that of the custom-designed stage by the lower boundary. The areas between the two curves represent design options that are "customized" to some degree. Because of the lower inert weight of the custom-designed modules, significantly shorter flight times are achievable in some instances.

A Centaur-class upper stage is adequate for Pioneer class orbiters. The Space Tug is required for Mariner class orbiters. Minimum flight times for Centaur-launched Pioneer orbiters range from 1600 to 1730 days for space-storable propellants, depending on whether a custom-designed or a multi-mission propulsion module is used. Use of the Space Tug as upper stage would reduce these flight times to about 1250 days. With earth-storable propellants only a custom-designed propulsion module (flight time 2100 days) can make use of a Centaur-class upper stage for Pioneer missions. With Space Tug as upper stage the earth-storable system achieves flight times of 1420 and 1490 days, respectively. Note that the maximum propellant load (P_L) of the multi-mission module determines a minimum flight time, indicated by the left boundary of the shaded



THE DATA REFLECT CONSERVATIVE PAYLOAD MASS ASSUMPTIONS (408 KG FOR PIONEER AND 680 KG FOR MARINER CLASS ORBITERS), SEE TEXT.

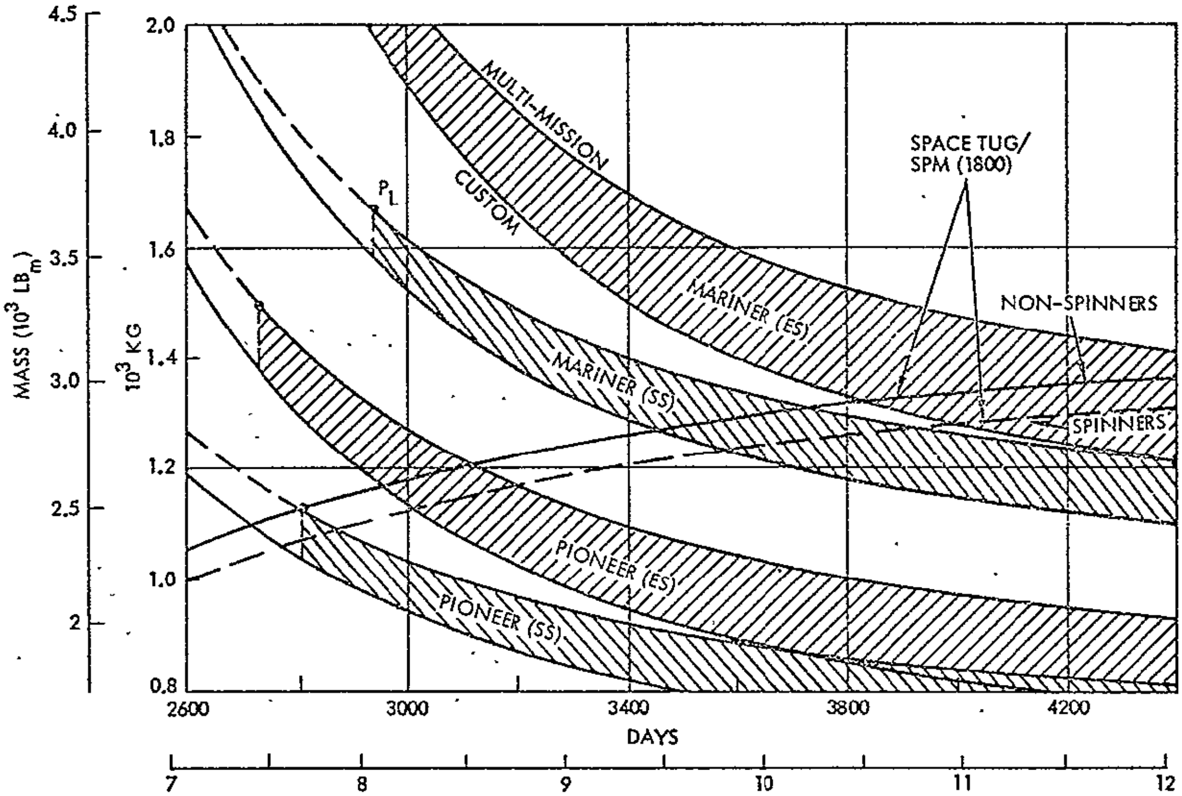
Figure 18. Performance of Pioneer and Mariner Saturn Orbiters (Updated Propulsion Module Inert Weight)

band, which in this case is about 25 days greater than the time given by the intersection of the spacecraft and launch vehicle characteristics; a 40 kg margin of launch vehicle performance remains that cannot be used to load more propellant.

Flight times for Mariner type payloads range from 1600 to 1700 days for space-storable and to 1900 days for earth-storable propulsion (custom designed module only). Use of an earth-storable multi-mission module is not feasible in the case of a Mariner payload, not even when launched by Space Tug/SPM (1800). These data do not reflect the use of the propulsion module for a C_3 augmentation maneuver (see below).

4.3.2 Uranus Orbiter

Figure 19 shows the corresponding performance plot for Uranus orbit missions. Only the Space Tug/SPM (1800) performance curve is shown in this graph since a Centaur-class upper stage would not be



THE DATA REFLECT CONSERVATIVE PAYLOAD MASS ASSUMPTIONS (408 KG FOR PIONEER AND 680 KG FOR MARINER CLASS ORBITERS),

Figure 19. Performance of Pioneer and Mariner Uranus Orbiters

adequate. Flight times range from about 2600 days for Pioneer type payloads to 3700 days for Mariner type payloads with space-storable propulsion. Mariner payloads with earth-storable propulsion require at least 3800 days with a custom-designed propulsion module and over 4400 days with a multi-mission module. A C_3-augmentation maneuver can provide some performance improvement (see below).

4.3.3 Effect of Launch Vehicle and Payload Mass Changes on Saturn and Uranus Mission Performance; Performance Summary

Performance data shown in Figures 18 and 19 were based on the assumption of fixed payload weights for Pioneer and Mariner type spacecraft. Actually, a range of payload weights for each spacecraft family was specified in the work statement. Figures 20 and 21 show performance plots for parametric payload weight variations (for multi-mission propulsion modules and space-storable propellants only) and additional launch vehicle upper stage candidates.

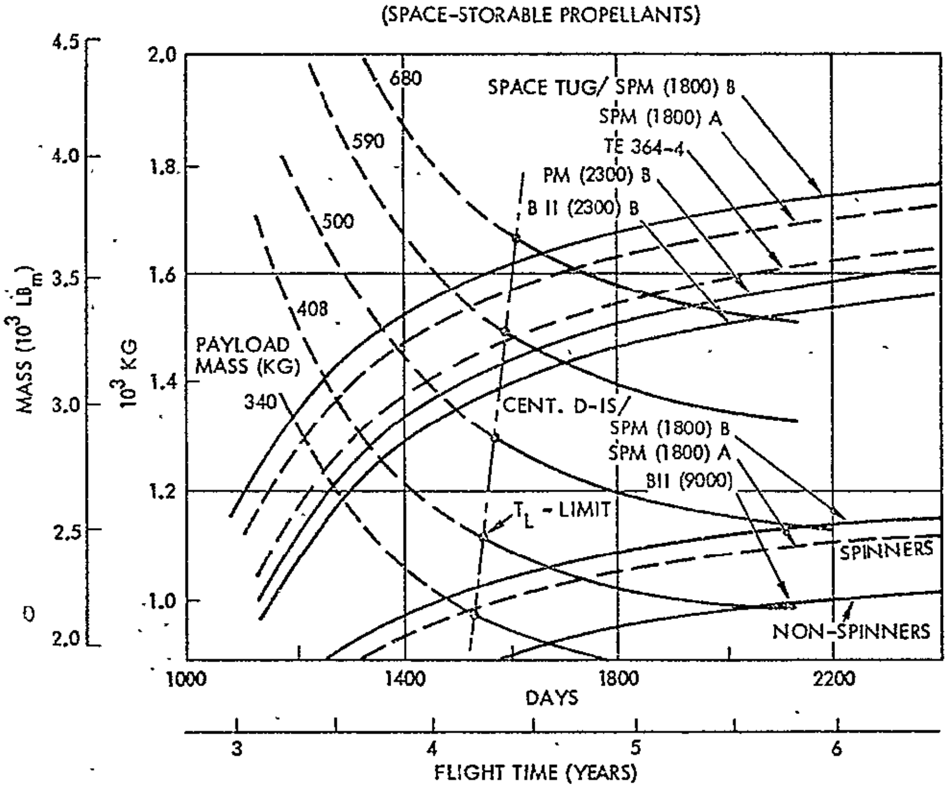


Figure 20. Saturn Orbiter Performance with Varying Payload Mass and Different Shuttle Upper Stage Candidates

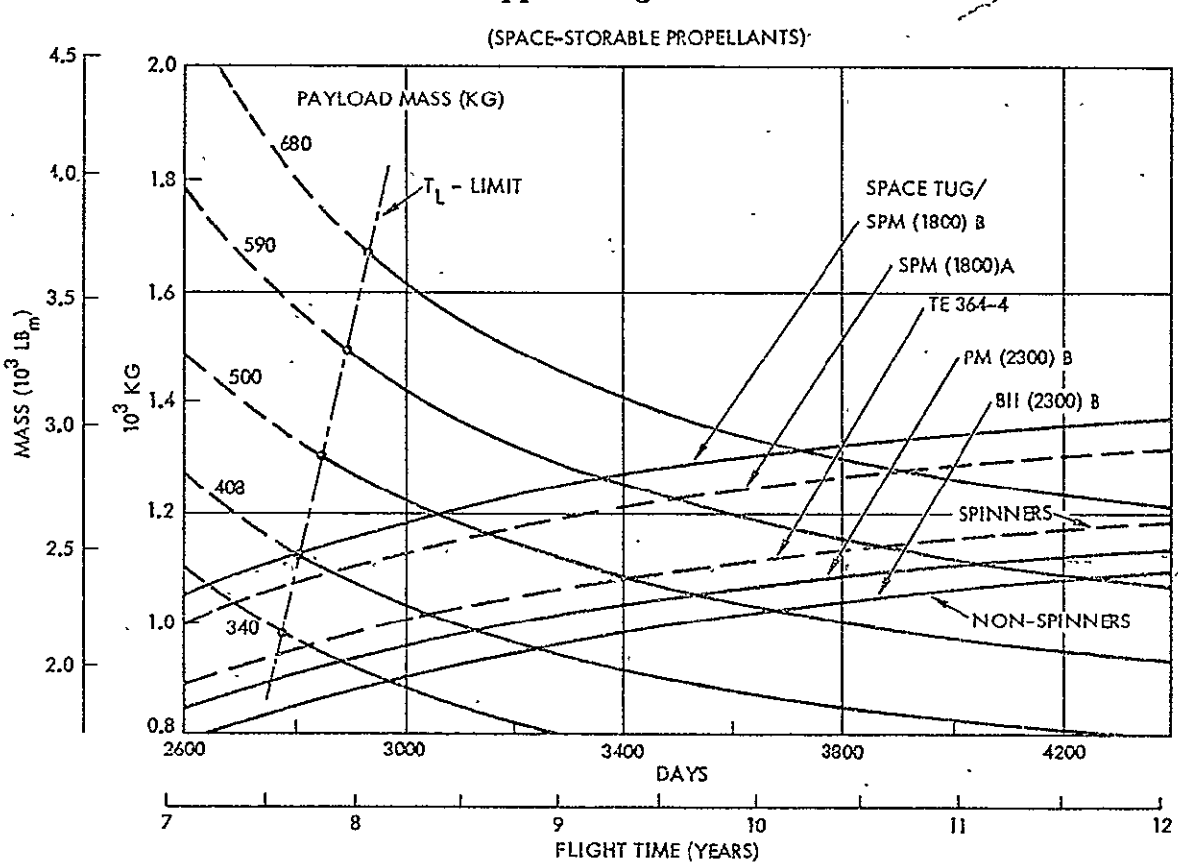


Figure 21. Uranus Orbiter Performance with Varying Payload Mass and Different Shuttle Upper Stage

The effect of C_3 augmentation is significant only in missions of long duration where the slope of the performance curves (Figures 18 through 21) levels out, and the location of their intersections (designating minimum flight time) becomes increasingly sensitive. Thus an augmentation of launch vehicle capability by 100 kilograms achievable by an expenditure of 200 to 300 kilograms of propellant weight, can produce flight time reductions of up to 200 days for the longest missions considered. However these results apply only in the case of space-storable propellants. Performance improvements by C_3 augmentation with earth-storable propellants are practically negligible.

Table 7 summarizes mission characteristics for representative Pioneer and Mariner class Saturn and Uranus orbiters and compares spacecraft and propellant masses and flight times with space- and earth-storable propellants. Figure 22 shows minimum flight times achievable in Saturn and Uranus orbit missions.

Results of the above performance comparison are summarized as follows:

1) Flight time reductions achievable by space-storable propellants for a given Shuttle upper stage are very significant (1 to 2 years), particularly for Mariner class payloads.

2) In missions with Pioneer type payloads the flight time reduction is not nearly as large, typically ranging from 0.5 to 1.0 year.

3) Only the use of space-storable propellants makes the multi-mission module concept, with its attendant cost economy, feasible and attractive in Mariner class missions.

4) In the case of Mariner missions to Saturn or Uranus an earth-storable multi-mission propulsion module would lead to unrealistically long flight times approaching those of Hohmann transfers.

5) C_3 augmentation, useful only with space-storable propellants, compensates for weight penalties inherent in the multi-mission module concept in some cases affording flight time reductions of 180 days or more in Uranus missions.

Table 7. Outer-Planet Orbiter Performance Summary

Payload	Propellants/ Module Type[1]	Trip Time[2] (days)	Propellant Capacity of MM Module kg (lb_m)	Propellant Mass kg (lb_m)	Inert Mass kg (lb_m)	Total Injected Mass kg (lb_m)
Saturn Orbiter						
Pioneer	ES/MM	1480[3]	894 (1971)	894[3] (1971)	617 (1360)	1511 (3332)
	CD	1400	-	910 (2007)	579 (1277)	1489 (3283)
	SS/MM	1720[4]	551 (1215)	480 (1058)	583 (1286)	1063 (2344)
	CD	1600[4]	-	480 (1058)	536 (1182)	1016 (2240)
Mariner	ES/MM	2160	1272 (2805)	930 (2051)	927 (2044)	1857 (4095)
	CD	1920	-	860 (1896)	846 (1865)	1706 (3762)
	SS/MM	1730	781 (1722)	725 (1599)	878 (1936)	1603 (3535)
	CD	1570	-	680 (1499)	828 (1826)	1508 (3325)
Uranus Orbiter						
Pioneer	ES/MM	3200	894 (1971)	570 (1257)	617 (1360)	1187 (2617)
	CD	3020	-	560 (1235)	544 (1200)	1104 (2434)
	SS/MM	2860	551 (1215)	510 (1125)	583 (1286)	1093 (2410)
	CD	2750	-	520 (1147)	540 (1191)	1060 (2337)
Mariner	ES/MM	4700	1272 (2805)	470 (1036)	927 (2044)	1397 (3080)
	CD	3840	-	575 (1268)	818 (1804)	1393 (3072)
	SS/MM	3750	781 (1722)	430 (948)	878 (1936)	1308 (2884)
	CD	3460	-	460 (1014)	806 (1777)	1266 (2792)

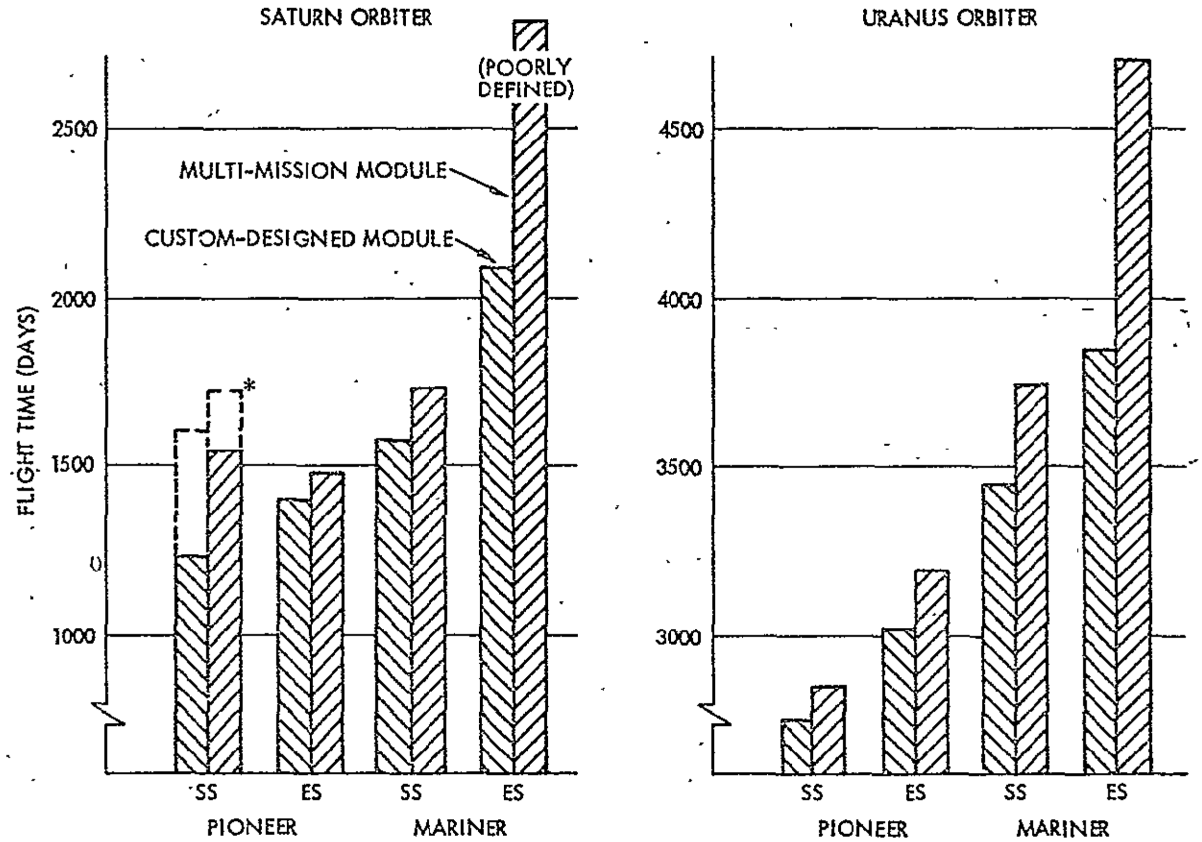
Notes:

[1] MM - Multi-mission; CD - custom-designed

[2] Does not reflect C_3 augmentation

[3] Dictated by maximum propellant capacity (50 kg launch weight margin)

[4] Launched by Shuttle/Centaur D-1S/SPM (1800)



*LAUNCHED BY SHUTTLE/CENTAUR D-1S/SPM (1800)

THE DATA REFLECT CONSERVATIVE PAYLOAD MASS ASSUMPTIONS (408 KG FOR PIONEER AND 680 KG FOR MARINER CLASS ORBITERS).

Figure 22. Flight Times for Saturn and Uranus Orbiters (Launch Vehicle Shuttle/Space Tug/SPM (1800) Except as Noted)

6) A lower-performance Shuttle upper stage, such as Centaur D-1S/SPM (1800), is adequate for Saturn missions by Pioneer class vehicles with space-storable propellants but with a flight time increase by more than 200 days compared to Space Tug/SPM (1800) launch.

4.3.4 Comet Mission Performance

Performance of the multi-mission module was evaluated for the seven specified comet rendezvous missions. Table 8 shows the results obtained for space-storable propulsion. Only one of these missions (Tempel 2) can be performed by a Pioneer or Mariner class spacecraft with a single propulsion module. Several others require tandem modules. Those with highest energy requirements cannot even be performed with two modules unless the payload mass is reduced.

Table 8. Characteristics of Comet Rendezvous Missions

Comet Mission	ΔV_{total}* (km/sec)	Number of Stages Required	Module Type	Propellant Weight Stage 2 kg (lb_m)	Propellant Weight Stage 1 kg (lb_m)	Initial Weight kg (lb_m)
1 Tempel 2	2.021	1	A	414 (913)	- -	997 (2198)
		1	B	623 (1374)	- -	1501 (3310)
2 Tempel 2	2.879	2	A	551 (1215)	159 (351)	1468 (3237)
		2	B	781 (1722)	285 (628)	2142 (4723)
3 Faye	3.760	2	A	551 (1215)	546 (1204)	1855 (4090)
		**	B	781 (1722)	845** (1863)	**
4 Kopff	2.521	2	A	551 (1215)	26 (57)	1335 (2944)
		2	B	781 (1722)	91 (201)	1948 (4295)
5 Perrine Mrkos	3.082	2	A	551 (1215)	240 (529)	1549 (3416)
		2	B	781 (1722)	403 (889)	2260 (4983)
6 Perrine Mrkos	4.062	**	A	551 (1215)	700** (1544)	**
		**	B	781 (1722)	1075** (2370)	**
7 Encke	4.110	**	A	551 (1215)	726** (1601)	**
		**	B	781 (1722)	1112** (2452)	**

*ΔV_{total} includes major midcourse (ΔV_2) and rendezvous maneuver (ΔV_3) plus 300 m/sec for guidance corrections and for excursions at comet. (For Encke only 200 m/sec of extra maneuver capability are allowable.)

**In these missions propellant requirements exceed tandem-stage capacity if payload mass of 408 (for Module A) and 680 (Module B) is assumed. Payload mass reduction of about 50 kg is required to make missions 3(B), 6(A and B), and 7(A) feasible.

5. DEVELOPMENT PLAN AND COST ASSESSMENT

5.1 PROGRAMMATIC CONSIDERATIONS

One of the principal objectives of the study was to assess the costs, both recurring and nonrecurring of the four multi-mission chemical propulsion modules as a function of the number of missions they might serve, and to estimate the total time required to develop and bring the stages to operational status.

Development of a propulsion module for either a custom design for a specific mission or a multi-mission stage will begin from the technical (state-of-the-art) and hardware basis in effect at the time. Components such as rocket engines, valves, controls and possibly, tankage may be adapted from other programs. However, the time frame and new handling and interface requirements for Shuttle launch imply that most components must be of new or modified design.

The new size, Shuttle launch requirements and long flight duration, imply a full propulsion development cycle, even if an existing engine were to be used.

For N_2O_4/MMH propulsion systems, a complete technology base exists on which to start development except possibly for some aspects related to long life, e.g., corrosion and material life.

LF_2/N_2H_4 systems, rocket propulsion hardware is in an advanced technology status with flight qualification of a system not yet accomplished, and a longer development cycle will be required than for N_2O_4/MMH. It is assumed that systematic technology development will continue to be followed by a development program aimed at mission applications such as those considered here.

The main incentive for a multi-mission stage development is cost saving. For a single mission, a custom-designed stage may have cost advantages, however slight, over a multi-mission stage mission. It also will have performance advantages because design and size are optimized for this mission. Custom-designed stages would be almost identical except for tank size and thrust level. Differences in development cost for custom and multi-mission stages then depend on 1) mission, 2) availability of hardware, and 3) type and amount of propellants.

Stages of N_2O_4 and MMH, and to a certain extent LF_2/N_2H_4, if developed soon, would undoubtedly utilize hardware adapted from the TRW MMBPS or JPL Mariner programs. In order to assess the costs for the multi-mission stages under the performance, reliability and safety requirements of the study, new propulsion stage costs were developed as an upper boundary.

In order to assess the multi-mission stage program cost relative to a minimum cost program, direct application of an existing stage, the TRW MMBPS, with only structural and thermal modifications and an unmodified, existing 94 lb_f engine was considered for comparison. Such a stage adaptation, suitable for a Pioneer Jupiter orbiter, launched from a Titan Centaur, was studied by TRW under NASA/Ames contract (Reference 9) and serves as reference.

Differences in the propulsion system hardware development items between the various stages are indicated by X's in the chart below:

	Module A		Module B	
	E.S. N_2O_4/MMH	S.S. LF_2/N_2H_4	E.S. N_2O_4/MMH	S.S. LF_2/N_2H_4
Gimballed engine			X	X
Bipropellant ACS	X			
ACS propellant in main tanks	X	X		
Capillary propellant acquisition (N_2H_4 only)		X		
Sun shade, deployable		X		
Cryogenic tank		X		X

It is expected that missions to the outer planets will be performed earlier than the Mercury missions. This places emphasis on the 900 Newton (200 lb_f) thruster development having a higher priority than 3600 Newton (800 lb_f) thrusters. (It should be noted that the 1971 Mariner orbiter used a 1300 Newton (300 lb_f) N_2O_4/MMH thruster with a specific impulse of 290 seconds).

5.2 COSTING GUIDELINES

Four sets of costs and time requirements are anticipated for the four stages to be presented and shown in Section 5.5.

- Four baseline multi-mission stages were defined: A-N_2O_4/MMH, A-F_2/N_2H_4, B-N_2O_4/MMH, and B-F_2/N_2H_4

- Stages sized and optimized for an ambitious mission; the 1988 Mercury Orbiter 735-day flight covered as a tandem application

- System performance of this configuration was assessed for the other missions

- A development start date of 1 January 1976, and a minimum cost development schedule were established. This means a fairly short schedule for the N_2O_4/MMH system and a longer one for the F_2/N_2H_4 system. This date is suggested to establish early fixed-cost and state-of-the-art benchmarks for the study

- Development costs generated on either/or (independent) basis not on the basis that two or more systems are started at once

- Cost estimates for the stages will be related to TRW past programs. Labor, materials, and overhead rate assumptions were mutually agreed upon with NASA as representing industry

- Costs were expressed in constant dollars based on a fixed date — 1 January 1975

- It was assumed that programs will be conducted in accordance with usual spacecraft propulsion development procedures

- Stage hardware is assumed delivered to the government prior to payload vehicle integration.

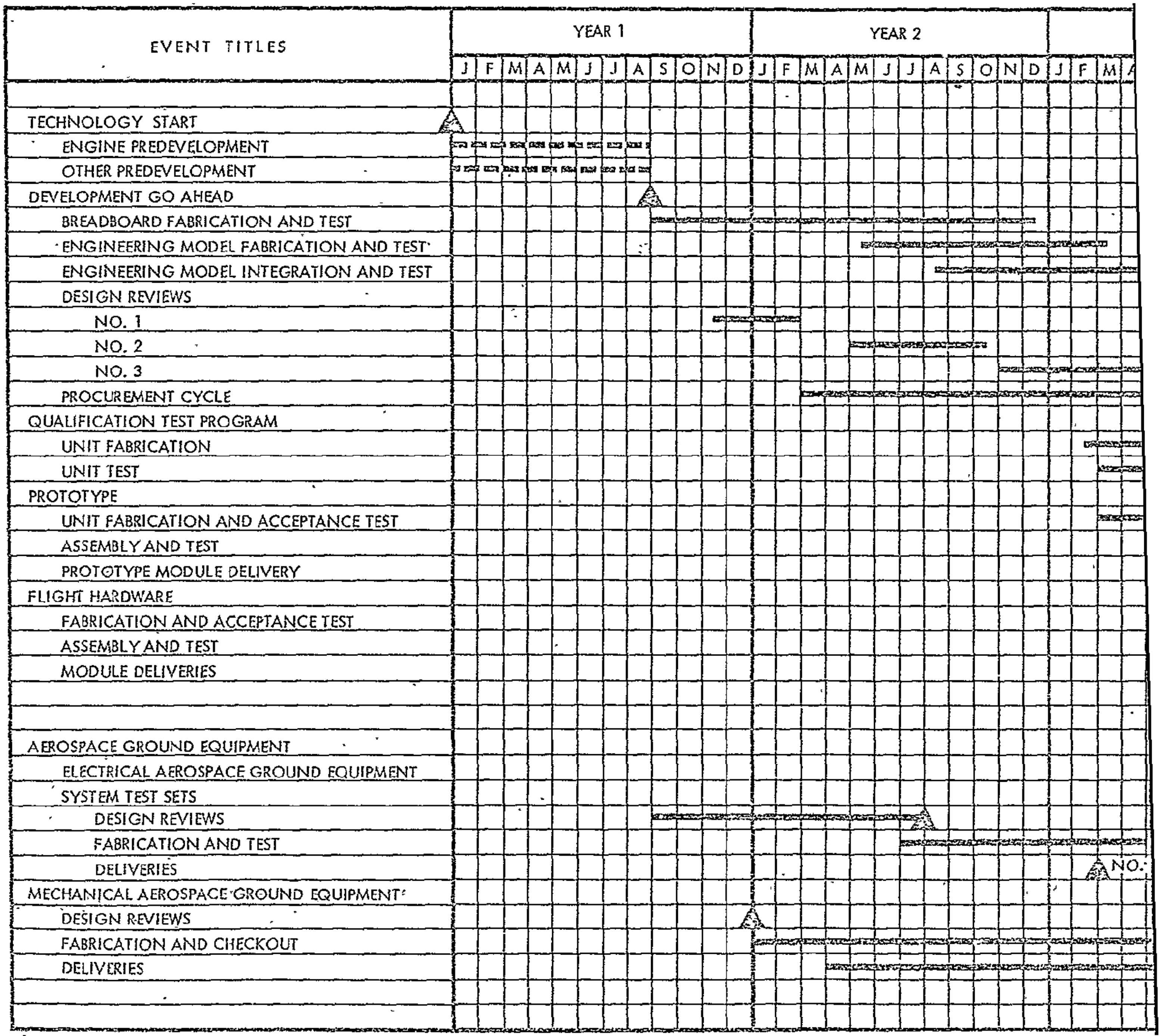
5.3 SUMMARY DEVELOPMENT SCHEDULE

5.3.1 N_2O_4/MMH Development Schedule

Figure 23 shows a typical development schedule for an N_2O_4/MMH propulsion stage. The first 8 months shown represent an optional 8-month technology cycle the purpose of which is to demonstrate the technology necessary to begin full development of a propulsion stage or module. If an existing engine can be used, the overall cycle can be shortened.

Time required from development go-ahead to flight module delivery is 30 to 44 months with 36 months being typical. If an engine predevelopment

FOLDOUT FRAME



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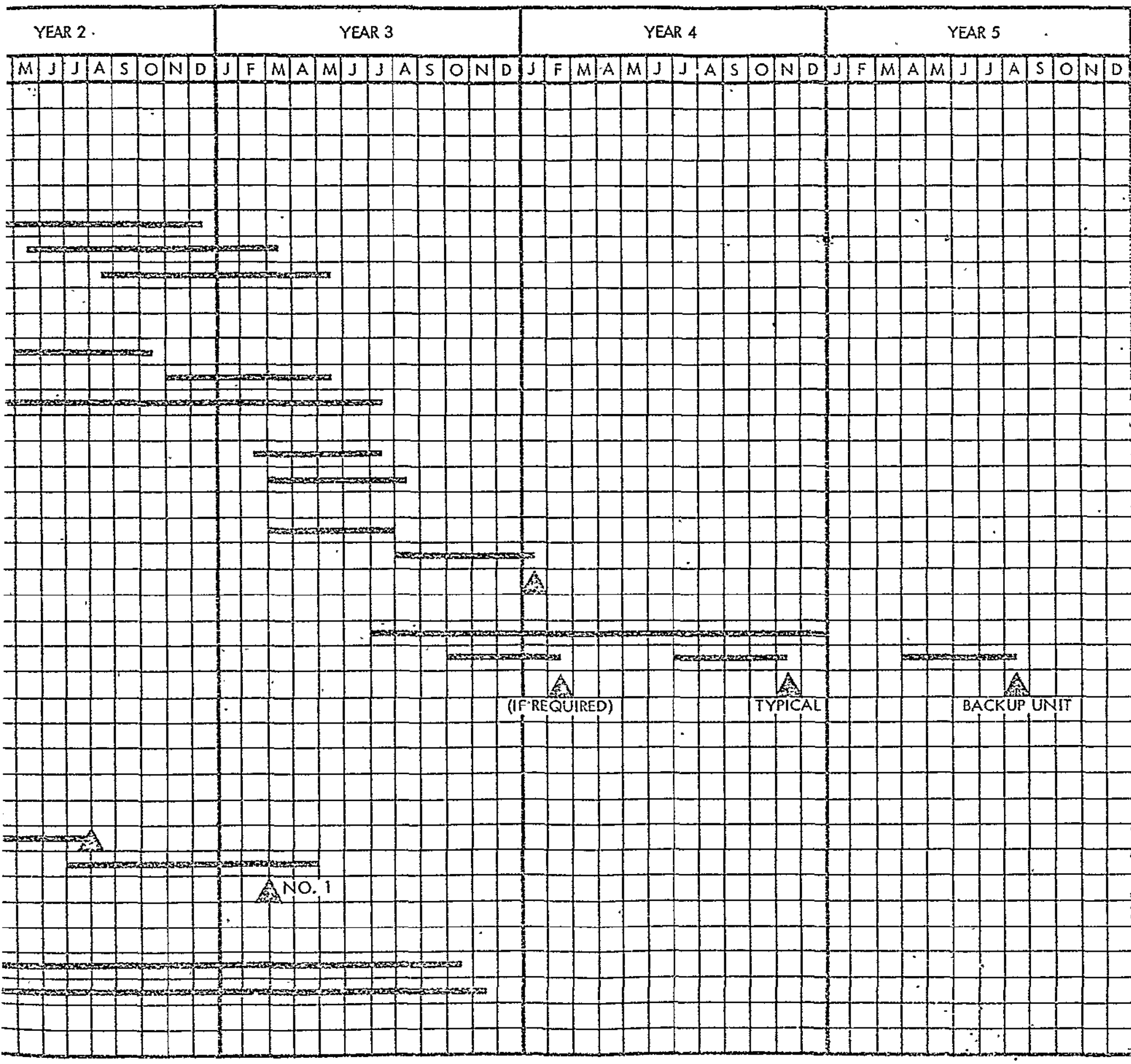

Figure 23. N_2O_4/MMH Development Milestone Schedule

is required then the cycle is 38 to 44 months. Aerospace Ground Equipment (AGE) deliveries can precede the flight hardware deliveries.

LF_2/N_2H_4 Development Schedule

Figure 24 shows the schedule anticipated for development of an LF_2/N_2H_4 propulsion system. This development schedule is similar to that for N_2O_4/MMH except that technology work is mandatory. An estimated 20 months of technology work on engine demonstration, valve technology for engine and propellant isolation valves, and materials and processes will be needed before a development comparable to the N_2O_4/MMH program could be started.

Duration of the F_2/N_2H_4 development program is thus approximately 50 to 58 months. This compares to approximately 38 to 44 months for N_2O_4/MMH system. These estimates are based on the assumption that no major technical difficulties are experienced.

Since technical risk is higher in the LF_2/N_2H_4 development program the longer technology period is recommended, so that technical questions can be resolved prior to full commitment.

Spacecraft development cycles can be even longer. An obvious conclusion is that if fluorine propulsion is needed in 4 or 5 years, the needed technology work should be instituted so that decisions can be made at an early date.

5.3.2 Custom-Designed Propulsion Module

N_2O_4/MMH Propulsion Modules

Custom-designed propulsion modules developed to meet the same performance, reliability, and safety requirements will be very similar in schedule and cost to a multi-mission module.

Spacecraft propulsion systems which might be adapted include the JPL Mariner with 440 kg capacity (970 lb_m in two tanks), TRW's MMBPS 600 kg (1300 lb_m) in four tanks), or JPL Viking 1408 kg (3097 lb_m in two tanks) capacity systems.

Orbiters based on these stages might be possible for the outer planet missions with Pioneer and Mariner class payloads, however either would require major repackaging.

EVENT TITLES	YEAR 1												YEAR 2													
	J	F	M	A	M	J	J	A	S	O	N	D	J	F	M	A	M	J	J	A	S	O	N	D	J	F
TECHNOLOGY START																										
ENGINE PREDEVELOPMENT/VALVE TECHNOLOGY/																										
MATERIALS AND PROCESSES																										
PREDEVELOPMENT VALVE AND MATERIALS																										
DEVELOPMENT GO AHEAD																										
BREADBOARD FABRICATION AND TEST																										
ENGINEERING MODEL FABRICATION AND TEST																										
ENGINEERING MODEL INTEGRATION AND TEST																										
DESIGN REVIEWS																										
NO. 1																										
NO. 2																										
NO. 3																										
PROCUREMENT CYCLE																										
QUALIFICATION TEST PROGRAM																										
UNIT FABRICATION																										
UNIT TEST																										
PROTOTYPE																										
UNIT FABRICATION AND ACCEPTANCE TEST																										
ASSEMBLY AND TEST																										
PROTOTYPE MODULE DELIVERY																										
FLIGHT HARDWARE																										
FABRICATION AND ACCEPTANCE TEST																										
ASSEMBLY AND TEST																										
MODULE DELIVERIES																										
AEROSPACE GROUND EQUIPMENT																										
ELECTRICAL AEROSPACE GROUND EQUIPMENT																										
SYSTEM TEST SETS																										
DESIGN REVIEWS																										
FABRICATION AND TEST																										
DELIVERIES																										
MECHANICAL AEROSPACE GROUND EQUIPMENT																										
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FABRICATION AND CHECKOUT																										
DELIVERIES																										

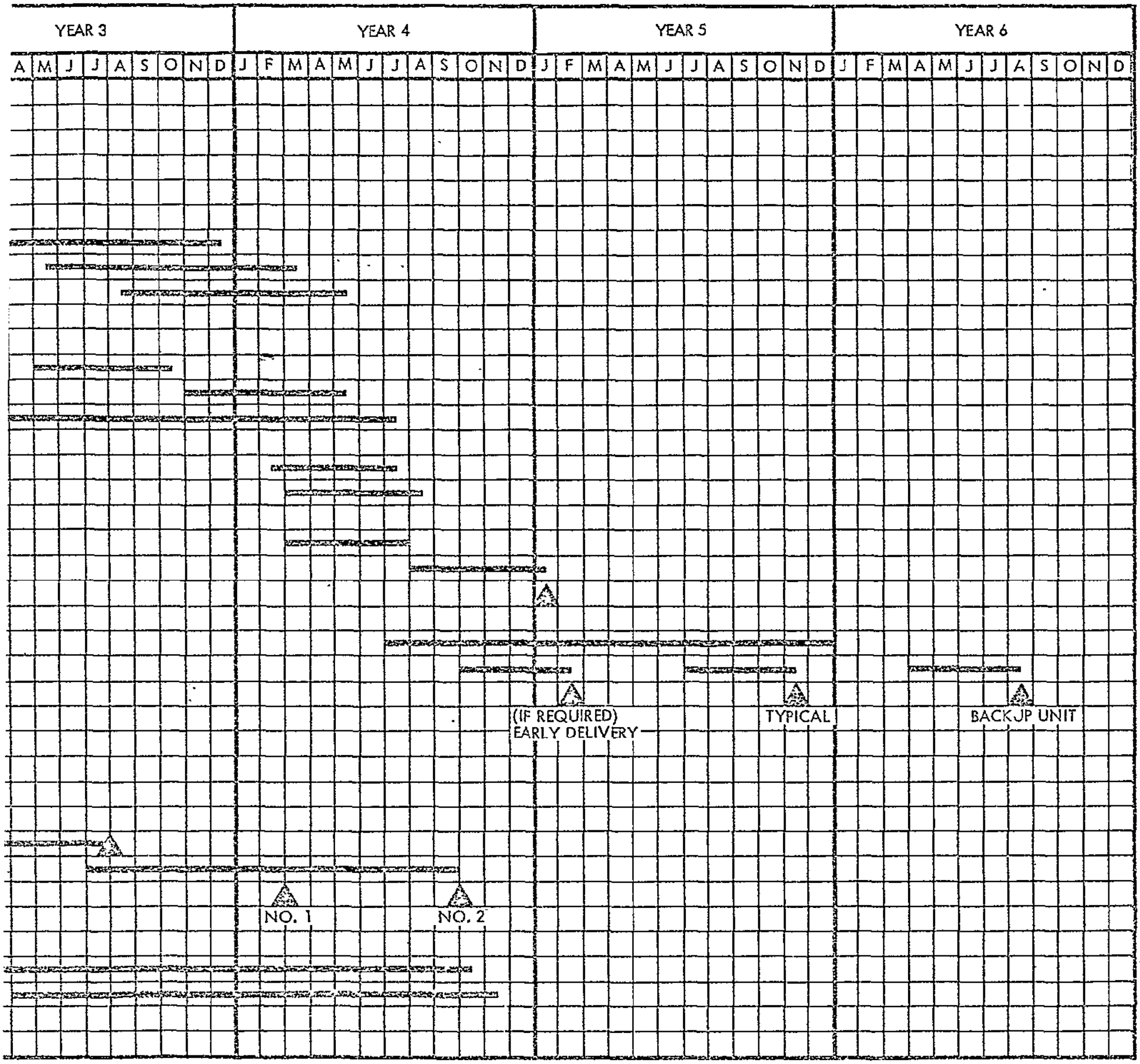

Figure 24. LF_2/N_2H_4 Development Milestone Schedule

The estimated schedule and cost for an outer-planet orbiter based on repackaging components from these systems is 24 to 30 months and approximately 1/3 the cost of an all new multi-mission module or all new custom-stage. Such a repackaged stage would have the following disadvantages:

1) Propellant expulsion bladders are of questionable reliability because of the long flight time

2) The system may not be suitable for Space Shuttle launch

3) It would not have as high a mass fraction or specific impulse.

Adaptation of any custom design to a new mission requiring a significant tank or structure change will require a lead time of at least 18 to 24 months to build, wring out, and qualify. Engine and control components can remain the same provided their original sizing is for the 800 lb_f level.

LF_2/N_2H_4 Propulsion Module

The first custom or multi-mission propulsion module to be qualified and flown using LF_2/N_2H_4 will incur significantly higher costs. However, costs of subsequent adaptation and requalification should approach those of the N_2O_4/MMH stages. No existing stages can be used for adaptation. However, some use of existing hardware on the N_2H_4 system may be possible.

5.4 DEVELOPMENT COST ESTIMATES

Table 9 shows the estimated nonrecurring and recurring costs for the systems considered.

The earth-storable cost data represent an all new system designed for Shuttle launch requirements.

There is approximately a $8.4 million cost difference between the F_2/N_2H_4 stage compared with the N_2O_4/MMH stage. Of this approximately $3 million consists of technology expected to be needed to assure a low risk development and $5.4 million is related to the additional engineering, hardware and test costs during development.

Table 9. System Cost Estimates

Cost Elements	Module A New Earth-Storable	Module A Space-Storable	Module B New Earth-Storable	Module B Space-Storable	Repackaged Existing Earth-Storable
NON-RECURRING COST					
1. Configuration with 800-lb$_f$ engine					
Predevelopment	450	2,996	450	2,996	
Module development	15,510	21,330	15,510	21,330	5,100
Sun shade		1,150			
ACS engine, bipropellant	1,640				
ACS monopropellant requal		250	250	250	
Gimbal actuators			300	300	
Total	17,600	25,726	16,510	24,876	
Alternate Configurations					
2. Module with existing 96-lb$_f$ engine	15,650	-	15,560	-	5,100
3. Module with 200-lb$_f$ engine	17,580	25,550	-	-	
4. Module with two engines (200- and 800-lb$_f$)	19,080	28,110	18,010	26,876	
RECURRING COST*					
Module system	1,170	1,470	1,170	1,470	920
Sun shade, acceptance		200			
10 bipropellant ACSE	273				
10 monopropellant ACSE		250			
16 monopropellant ACSE			400	400	
Gimbal actuators			100	100	
Total	1,443	1,920	1,670	1,970	1,170

*Based on a production run of 10 stages. Costs shown are for one stage

Module A and B are not greatly different in physical size and have the same maximum thrust level. Except for the sun shade on the space-storable Module A and fewer thrusters on Module A they are similar in complexity. The only significant differences are in tank size. Lines, valves, and engines could, and most likely would, be identical (and sized for 800 lb$_f$). Since there is so little difference in equipment and complexity, development costs are very similar.

Shuttle-launched custom stages built to the same performance, reliability and safety specifications as a multi-mission module are very likely to have the same nonrecurring cost. The difference in tank

and test costs is insignificant within the accuracy of estimating. Recurring cost would, of course, be much higher for custom-designed stages. For custom-designed stages, system development, normally a non-recurring cost would continue to be a recurring cost. Each new configuration would require \$5 to 7 million to modify and requality assuming a single hardware set for requalification.

The repackaged existing stage is shown for comparison and represents the cost of repackaging the components of an existing N_2O_4/MMH propulsion system in a new structure, with a monopropellant ACS system and requalifying it for flight on Titan/Centaur (not the Shuttle) as mentioned previously. No facilities costs were included.

Figure 25 shows the resulting cumulative costs of multi-mission and custom-designed stage procurement for Module A with earth-storable (left) and space-storable propellants (right) as function of the number of flights. The bar graphs illustrate the rapid accumulation of higher costs in the cost of individual (custom-designed) stages amounting to differences of \$17.8 million and \$20.4 million, respectively, assuming 6 flights.

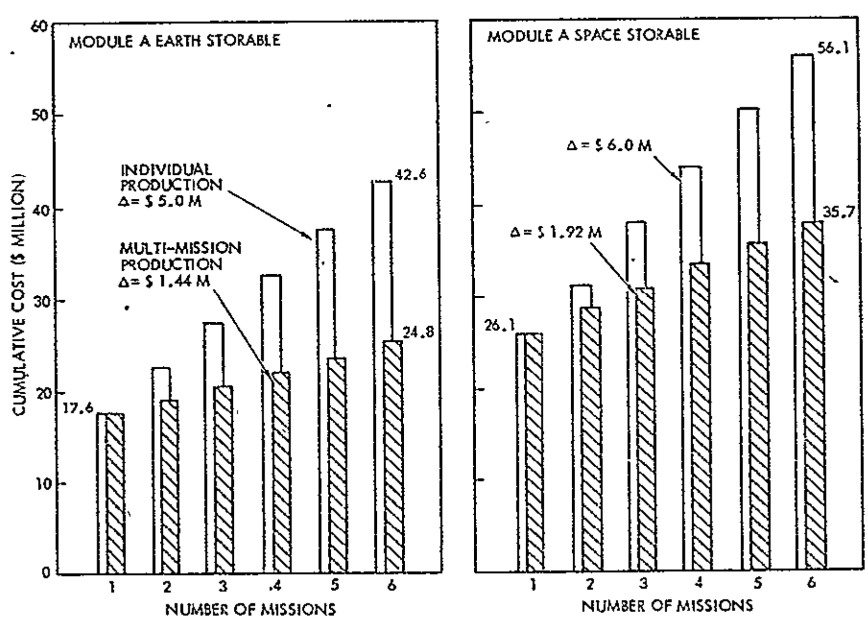

Figure 25. Comparison of Cumulative Cost of Individually Produced Propulsion Modules Vs. Multi-Mission Module Production

6. NEW TECHNOLOGY REQUIREMENTS AND COST EFFECTIVENESS

6.1 TECHNOLOGY ADVANCES REQUIRED FOR MULTI-MISSION MODULE

A major study objective was identifying new technology items necessary or desirable to meet the performance objectives of the multi-mission module most effectively, and assessing their cost effectiveness.

New technology is necessary to accomplish some of the specified missions launched by the Shuttle and Shuttle upper stages. There is a critical need to conserve injected weight in some of the missions. The higher-performance, space-storable F_2/N_2H_4 propellant combination is needed to perform many of the Shuttle-launched missions with the IUS or even with the Space Tug. At the large propellant weight-to-inert weight exchange ratios typical for this mission range, a 100-kilogram saving in inert weight can yield up to 500-kilogram savings in total injected weight. Thus, the cost of propulsion technology development avoids even more costly payload accommodation problems.

In this context, time is an important element. The multi-mission module flight programs are intended for the mid-1980's with the advent of Space Shuttle. Technology advances are achievable if development starts immediately. Available lead time must be factored into the technology-versus-cost assessment and could become a sensitive factor if underestimated.

The new technologies most necessary, and the benefits to be gained, are summarized in Table 10.

6.2 ESTIMATED NEW TECHNOLOGY EVOLUTION - SCHEDULE AND MILESTONES

The evolutionary schedule for N_2O_4/MMH depends on funding plans and normal lead times for hardware and engineering. For the LF_2/N_2H_4 combination, approximately $3,000,000 for predevelopment is required to allow full scale development starting in fiscal year 1978. Thus, no fluorine flights could occur before mid-1981 calendar year (or, with more conservative spacecraft lead time estimates, not before 1982).

Table 10. Summary of New Technology Requirements and Suggested Innovations

	Improvement	Category	Cost, $M
New Technology			
1. Deployable sun shade*		Essential for Module A for Mercury	1.150
2. LF_2/N_2H_4 800 lb_f engine technology	Demonstrate feasibility at I_{sp} = 370 + sec.	Essential for LF_2 Mercury missions	0.993
3. LF_2/N_2H_4 200 lb_f engine technology	Demonstrate feasibility at I_{sp} = 370 + sec.	Essential for LF_2 Mercury missions	0.981
4. Long-life isolation valves for LF_2		Essential for LF_2/N_2H_4 systems	1.000
5. LF_2 materials and processes technology	Determine compatibility and passivation	Essential for LF_2/N_2H_4 systems	1.000
6. Improved I_{sp} 200 lb_f N_2O_4/MMH engine	Demonstrate increase in I_{sp} state of the art to approximately 310 sec.	Beneficial	1.930
7. Improved I_{sp} 800 lb_f N_2O_4/MMH engine	Demonstrate increase in I_{sp} SOTA to approximately 310 sec.	Beneficial	1.950
8. Development of 2 lb_f N_2O_4/MMH ACS thrusters	Reduce ACS propellant approximately 1/3 for Module A with earth-storables	Beneficial	1.650
Suggested Innovations			
9. Technology of N_2O_4/N_2H_4 engine - 200 lb_f alternative to 6.	Allows common tanking of ACS and main propellant	Beneficial	2.430
10. Technology of N_2O_4/N_2H_4 engine - 800 lb_f alternative to 7.	Allows common tanking of ACS and main propellant	Beneficial	2.450
11. Development of 2 lb_f N_2O_4/MMH bimodal engine	Allows common tanking of ACS and main propellant	Beneficial	3.000
12. Technology of N_2O_4/N_2H_4 bimodal engine	Allows common tanking of ACS and main propellant	Beneficial	3.000

*Heat pipe would reduce cost to $0.2 million

Figure 26 includes the schedule for repackaging existing components into a structural configuration similar to the earth-storable multi-mission module for launch on Titan/Centaur (see TRW's Pioneer Jupiter Orbiter Study, Reference 9). This stage could be ready by early 1978. Figure 26 also shows mileposts for a scenario of technological evolution of both types of propulsion systems, assuming prompt and adequate funding without undue haste in the conduct of the programs.

6.3 QUALITATIVE COST-BENEFIT OF SPACE-STORABLE PROPULSION

Instead of making a quantitative cost-benefit assessment, overall system performance improvements made possible by advanced propulsion technology were considered in a qualitative manner. A recent JPL study (Reference 13) evaluated cost-benefits accruing from increased ΔV capability in Jupiter and Saturn orbiter missions. An increment of scientific value is gain by (1) extension of time in orbit, and (2) additional orbital maneuvers permitting observation of phenomena not observable in the preceding orbital phase. For example, in the Jupiter orbiter mission an initial ΔV of 1375 m/sec is required to establish the orbit and to permit continuation in the initial orbit for several years at a fixed rate of increase of scientific value per year. New maneuvers raising the total ΔV requirement to 2750 m/sec increase the growth rate of scientific value compared to a mission in which none of these maneuvers are performed.

The following criteria have been included in the qualitative cost benefit assessment:

- Payload mass increase
- Flight time reduction, with resultant increase in reliability and mission cost reduction
- Mission flexibility improvement, e.g., increase of the launch window
- Improved planetary exploration strategy permitting scientific results to be taken into account prior to making mission profile changes ("adaptive" mission strategy)

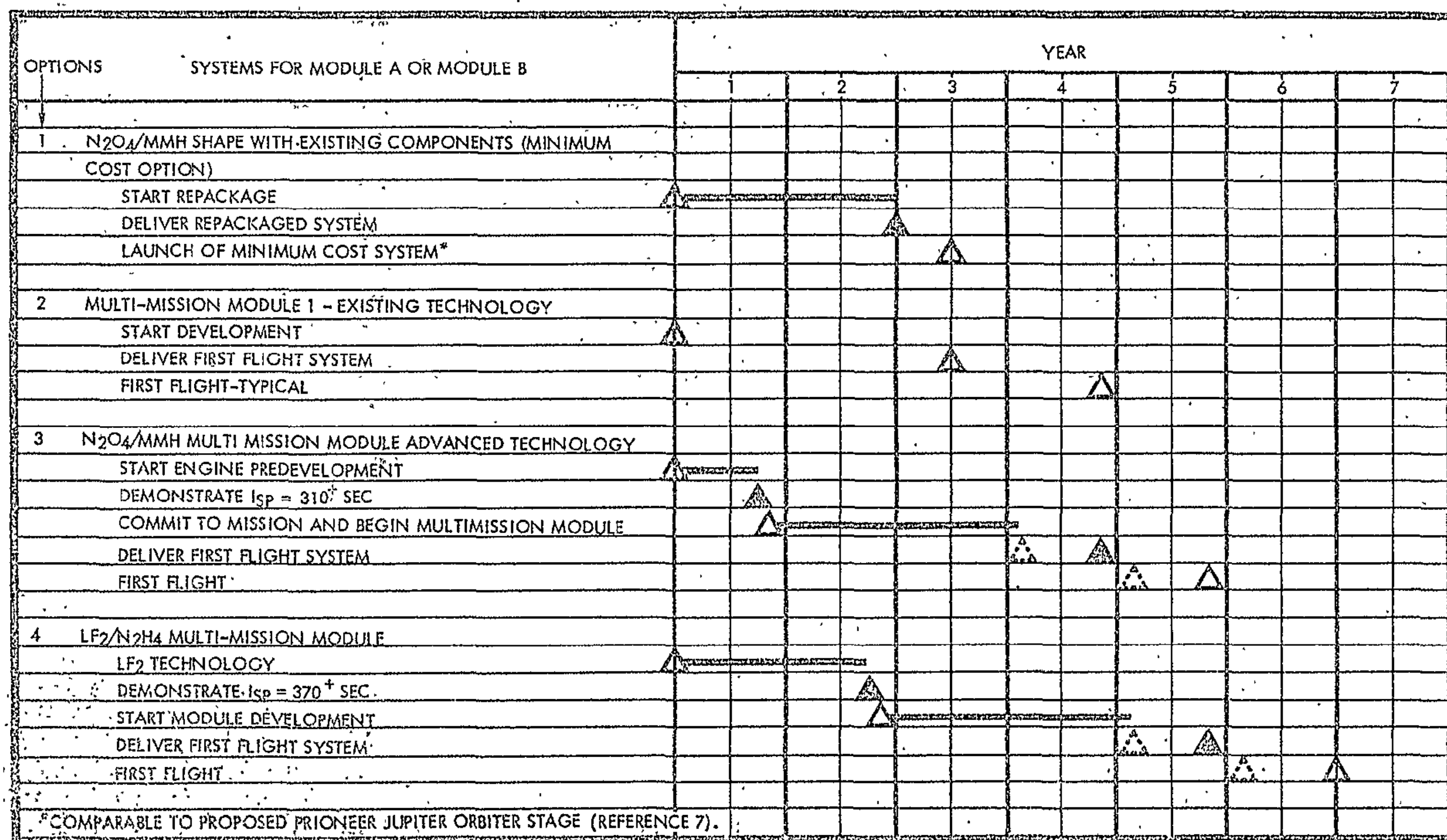

Figure 26. Availability Schedule of Propulsion Module Types

- Increased probability of success by permitting maneuvers to escape hazardous environmental conditions (e.g., change in periapsis altitude to avoid high particle flux, high thermal flux, or possible early impact on the planet surface)

- Reliability increase by adding weight for redundancy

- Payload cost reduction by relaxing weight and size constraints

- Reduction of the booster cost through lower launch vehicle performance requirements

- Achievability of missions that would not be feasible without the advanced propulsion technology.

Increased payload mass is a primary concern in mission planning and includes in part some of the other items listed above, such as added redundancy weight. Payload mass increase is inherent in the ability to perform planetary exploration by Mariner class rather than only Pioneer class orbiters with the possibility of accommodating more sophisticated instruments, such as higher resolution image systems, on the non-spinning spacecraft.

Payload mass increase also implies cost reductions by permitting adaptation of existing subsystems and/or scientific instruments without costly redesign.

Another possible benefit of payload mass increase is the ability to carry a planetary entry probe to the target planet. The addition of the entry probe means primarily added takeoff weight, not added inert weight during the orbital entry maneuver. However, it also requires additional onboard equipment such as relay communication system, and other probe support hardware. Mounting of the entry probe does not necessarily increase the height of the flight spacecraft. In the nonspinning configuration one or several probes could be mounted off-center without imposing mass distribution problems.

Greater mission flexibility is of particular value in connection with the use of the Shuttle orbiter as launch vehicle. An increase in launch window duration made possible by increased spacecraft propulsion capability will make the tight turnaround schedule between Shuttle flights a less severe constraint on launch operations.

Adaptive strategy of planetary exploration is a matter of increasing interest to mission planners and scientific experimenters. In all missions being considered the physical environment and potential hazards existing at the target planet or comet are largely unknown at the start of the mission.

Increased maneuvering capability facilitates successive orbit modifications to maximize scientific data yield. Repeated satellite swingby maneuvers are facilitated, which in turn provide additional orbit modification options.

In comet rendezvous mission maneuver requirements to explore the comet more fully after establishing rendezvous are quite modest, typically of 100 to 200 m/sec, depending on the comet, the length of time of stay with the comet, and range of excursions to be performed. Previously these missions were believed to be the domain of solar electric propulsion and have been awaiting the advent of that technology. As shown by the performance assessment, a wide range of possible comet rendezvous missions can be performed and thus the cost effectiveness of introducing advanced chemical propulsion is greatly increased.

Table 11 lists advantages achievable by using space-storable instead of earth-storable propellants versus specific benefits accruing in terms of scientific mission yield, mission success probability, cost reduction and program management factors. These factors are given a tentative value ranking, and scores of maximum benefits are indicated for each category (by circled figures).

6.4 PERFORMANCE, COST AND RISK CONSIDERATIONS

Cost-effectiveness analysis must take into account three principal criteria from which a figure of merit can be derived, namely performance, cost and risk.

The performance criterion includes such factors as payload capability, flight time reduction, and extra maneuvering capability that will enhance scientific mission yield.

The cost criterion includes cost savings due to improved design approaches, simpler test procedures, etc., and reduced mission time.

Table 11. Rating of Advantages Achieved by Space-Storable Propulsion

Categories of Advantages Achieved / Specific Benefits →	Greater Science Returns	Cost Reductions: Less Development of Payload Instruments	Cost Reductions: Lower Spacecraft and System Cost	Cost Reductions: Lower Shuttle Utilization Cost	Cost Reductions: Lower Upper Stage Performance Required	Cost Reductions: Lower Mission Support Cost	Greater Mission Success Probability	Programmatic Advantages	Score
Payload increase	x x x	x x[1]	x[1]		x		x[2]	x[3]	9
Missions made feasible (Uranus, comets)	x x			x[4]	x			x	5
More missions achievable by multi-mission module			x x					x	3
Flight time reduction						x x	x	x	4
Launch window increase				x x			x	x	4
Increased ΔV capability/ adaptive mission profile (e.g., satellite encounters)	x x	x	x[5]				x x[5]		6
Hazard avoidance at destination (extra maneuvers)			x				x		2
Score	7	3	5	3	2	2	6	5	

Notes:

[1]Relaxed weight constraints on components

[2]Redundancy weight added for greater reliability

[3]Adaptation of existing hardware to other missions

[4]Additional Shuttle traffic

[5]Higher ΔV capability implies simpler navigation

The latter aspect of cost reduction can be very significant. For example, in concurrent JPL studies typical mission cost per year of transit is assumed as \$6 to 7 million, and cost per year in orbit (which requires more intensive ground operations and support) as large as \$13 million.*

The risk factor includes development risk, mission success (or failure) probability, and safety considerations, especially those involving ground handling and launch of the system by the Shuttle orbiter. It interacts with performance characteristics since higher payload potential implies a greater redundancy weight allowance, and lower flight time implies higher success probability, as discussed in the preceding section.

For purposes of cost-benefit analysis the cost of technology improvement is accounted for as a separate item from the cost reduction achieved by this investment.

Table 12 lists major items of technology improvement identified in this study and cost estimates for this improvement, and assesses the benefits in each of the three categories (performance, risk reduction, and cost reduction) in matrix form. Only rough estimates of the benefit in terms of percentage improvement are given. Further study would be required to establish more detailed estimates. These data are then used to determine an estimated cost effectiveness ratio, defined as the sum of the three benefits, divided by the respective technology cost increments also in percent, viz.,

$$CE = \frac{\frac{\Delta P}{P_o} + \frac{|\Delta R|}{R_o} + \frac{|\Delta C|}{C_o}}{\frac{\Delta C_T}{C_o}}$$

Each contribution is assumed to carry an equal weighting factor.

The results show that the development of space-storable propulsion technology, additional development of F_2 safety provisions and materials technology, and longer-life ACS thrusters score high on this scale. Development of the heat pipe approach for LF_2 tank thermal control in the Pioneer Mercury orbiter mission, although a specialized requirement, stands out as having the highest cost effectiveness.

*Informal communication from R. Chase, JPL

Table 12. Technology Benefit Analysis

Technology Improvement Item	Estimated Cost of Improvement, C_T		Estimated Benefits (percent)			Cost-Effectiveness Ratio CE	Priority of Development
	$M	%[a]	Performance $\Delta P/P_o$	Reduced Risk $\lvert\Delta R\rvert P_o$	Reduced Cost $\lvert\Delta C\rvert/C_o$		
1. Development of space-storable propulsion system (see Table 8-1)	8	16	30 to 100	10 to 20	10 to 20	1.9 to 8.8	High
2. Improved materials technology (e.g., LF_2 compatibility)[b]	3	6	10	10	10	5	High
3. Additional development of safety provisions for LF_2 ground handling and Shuttle launch[b]	2	4	-	10 to 20	-	2.5 to 5	High
4. Increased specific impulse of earth-storable propulsion systems (see Table 9-4)	2	4	10 to 15	-	-	2.5 to 3.8	Medium
5. Centrifugally actuated heat pipe for LF_2 tanks	0.2	0.7[c]	-	5	3[c]	11.4	High[c]
6. Bipropellant ACS thrusters in 9 to 20 N (2 to 5 lb_f) range (see Table 9-4)[d]	1.6	3.2	5	5	3	4.1	Medium
7. Longer life ACS thrusters	1	2	-	10	-	5	High[e]
8. Improved long-life reliability design techniques	2	4	5	10 to 20	5	5 to 7.5	Medium

[a] Assumes total flight spacecraft cost of $50M (average between Pioneer and Mariner type missions) as reference

[b] These items for additional technology work, over minimum requirements subsumed under Item 1

[c] Required only for Pioneer Mercury orbiter. Reflects a lower reference cost ($30M) than other entries.

[d] Recognizes earlier bipropellant ACS thruster development by Aerojet Liquid Rocket Company (Reference 35)

[e] Required particularly for Mariner outer-planet orbiters

7. CONCLUSIONS AND RECOMMENDATIONS

7.1 DEVELOPMENT OF MULTI-MISSION PROPULSION MODULES FOR PLANETARY ORBITERS

The following principal conclusions regarding the advisability of development of multi-mission propulsion modules are drawn from results of this study:

- Development of a multi-mission propulsion module even for only two of the specified missions rather than custom-designed stages involves lower overall costs.

- Performance advantages of multi-mission space-storable systems over corresponding earth-storable systems are significant and include not only spacecraft gross weight savings, but shorter trip times to distant targets, greater mission flexibility and scientific yield, and lower launch-vehicle capability requirements.

- With larger, more sophisticated payloads (Mariner spacecraft) space-storable propellants are essential if all the missions in the specified set are to be performed. Some of the missions (Uranus orbiter) cannot be achieved within practical time limits with the use of earth-storable propellants.

- With a lower payload weight (Pioneer spacecraft) all missions could be performed with earth-storable propellants, although less satisfactorily than with space-storable propellants.

- Cost-benefit advantages overwhelmingly favor space-storable over earth-storable propellants for multi-mission propulsion modules.

- The estimated development cost of a space-storable multi-mission module exceeds that of earth-storable modules by less than $10M.

These conclusions are based to a large extent on including the three high energetic planetary orbit missions, namely, Mercury, Saturn, and Uranus in the mission set postulated for multi-mission application. Should the Uranus orbiter be given a lower priority, or be eliminated, the strength of the argument for space-storable propellants would be diminished to some extent.

Seven comet missions in the late 1980's and early 1990's were included, but only as secondary objectives. Most of these comet missions

can be performed if space-storable propellants are available, but some only by using two propulsion modules in tandem, as in the case of the Mariner orbiter. Making the multi-mission module as small as possible consistent with reduced flight times to the outer planets and efficient orbit insertion limits the propellant capacity. Thus, in the tradeoff between planetary orbiter performance and comet-rendezvous mission feasibility, the former was favored in the design approach.

7.2 ACCOMMODATION BY AND INTERFACES WITH PAYLOAD VEHICLES

The overall systems viewpoint requires that the multi-mission module concept be implemented without imposing difficult and/or costly interface and accommodation requirements on the payload. The cost benefits achievable by the multi-mission module would be partially defeated if major redesign of the existing Pioneer and Mariner spacecraft were necessary. These constraints were taken into account, but not all tradeoffs for a cost-effective overall systems approach were possible within the framework of this study.

Future work should consider detailed structure and performance aspects of Pioneer and Mariner spacecraft, especially the problem of structural reinforcement against high thrust accelerations. It is to be noted that even with a custom-designed propulsion module configuration some payload vehicle modifications are inevitable, e.g., reinforcement of the solar panels and the change in sun shield location to accommodate the orbit injection pointing requirements in the case of the Mariner Mercury orbiter. Therefore, only part of the added cost and weight penalties associated with such payload vehicle changes are chargeable to the multi-mission propulsion module concept when comparing its effectiveness with that of the custom-designed propulsion module.

7.3 ACCOMMODATION BY AND INTERFACES WITH THE SHUTTLE AND UPPER STAGES

The selected multi-mission stage design concepts satisfy size, weight, and structural constraints imposed by Shuttle launch. Safety requirements involving the use of fluorinated propellants were reflected in the design approach. Other handling, operational, and interface aspects were adapted from concurrent JPL studies.

7.4 TECHNICAL INNOVATIONS

Innovations identified and investigated in the course of the study include:

- The use of double-walled propellant tanks for greater safety and added micrometeoroid protection
- Use of a spin-deployed sun shade for Pioneer Mercury orbiters. Sun shade stowage, deployment, and dynamic properties were investigated but still require further study
- Use of a spin-actuated heat pipe for LF_2 tank thermal control in the Pioneer Mercury orbiter. This heat pipe concept would reduce size and complexity of the deployed sun shade. A fixed sun shade may, in fact, be adequate with this thermal control approach. This concept also is recommended for further study.

7.5 PROPULSION SYSTEM TECHNOLOGY AREAS RECOMMENDED FOR FURTHER STUDY

In addition to the novel concepts listed above, the following propulsion technology areas are recommended for further study and research, particularly in relation to reliability improvement:

1) Propulsion-system design for optimum redundancy. Methods for achieving at least partial mission success in the event of component failures.

2) Propellant corrosion (stress corrosion) of materials used in tanks and valves, including test and verification approaches.

3) Development of very high-reliability ACS thruster valves.

4) All aspects of system safety engineering, especially for LF_2/N_2H_4 (with emphasis on Shuttle launch requirements).

5) Design and utilization of double-wall tanks as related to system reliability aspects and safety during transport by the Shuttle orbiter.

6) LF_2/N_2H_4 engine technology, especially problems of non-equilibrium gas flow, combustion, pressure distribution, and cooling.

7) Design and applicability of N_2O_4/N_2H_4 engine.

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